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RDTE PROJECT NO.
USAAVSCOM PROJECT NO. 66-04
USAASTA PROJECT NO. 66-04

ENGINEERING FLIGHT TEST

YUH-1H HELICOPTER

PHASE D

(LIMITED)

FINAL REPORT

FLOYD DOMINICK
PROJECT ENGINEER

EMERY E. NELSON
LTC, US ARMY RET.
PROJECT PILOT

NOVEMBER 1970

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US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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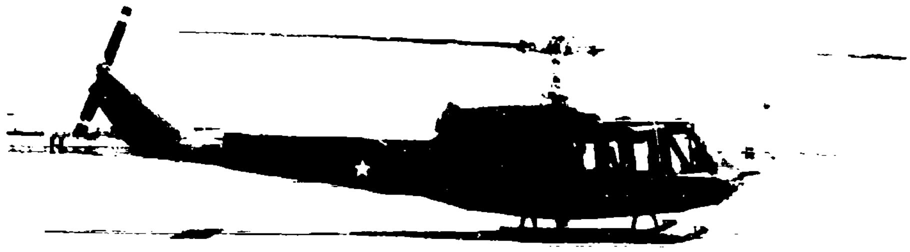

ABSTRACT

This report presents the results of a limited Phase D Engineering Flight Test of the YUH-1H helicopter equipped with the Lycoming T53-L-13 engine. The objectives of the test were to define the increased performance, to evaluate the engine characteristics and to investigate the flying qualities resulting from the expanded flight envelope. One hundred and fifteen data flights were conducted from December 1966 to August 1967 accounting for 98 productive flight hours. Testing was conducted in California at Edwards Air Force Base and at auxiliary test sites at Bakersfield and Bishop. Helicopter performance was quantitatively defined for hover, takeoff, climb and level flight. Engine characteristics and helicopter flying qualities were determined quantitatively at specific flight conditions and were qualitatively investigated throughout the operating flight envelope. The performance of the UH-1D was significantly improved by the installation of the higher powered T53-L-13 engine which resulted in an increased weight, temperature and altitude flight envelope. The T53-L-13 engine demonstrated good dynamic characteristics except for instability at topping power and poor static characteristics. In general, the helicopter flying qualities were acceptable in forward flight. During low-speed flight and hover, the left directional control margin was inadequate. At a forward center of gravity, the helicopter has insufficient aft longitudinal control. To safely use the increased flight envelope, the control deficiencies must be corrected.

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INTRODUCTION

BACKGROUND

1. In April and May 1963, the airframe contractor submitted proposals to install and evaluate a T53-L-13 engine as an alternate power plant for the UH-1D helicopter. The engine manufacturer's test data indicated that the UH-1D hover capability, climb performance and engine acceleration would be improved with the alternate power plant. In January 1965, the US Army Aviation Systems Test Activity (USAASTA), conducted a limited engineering flight test of the YT53-L-13 engine installed in the UH-1D helicopter. The results of this test were reported in reference 1, appendix I. The decision to install the T53-L-13 engine in future production UH-1D helicopters resulted in a requirement (ref 2) for conducting the tests outlined in reference 3. The UH-1D helicopter with the T53-L-13 engine installed was redesignated the UH-1H.

TEST OBJECTIVES

2. The overall objectives were as follows:

a. To define the increased performance provided the UH-1D by the installation of the T53-L-13 engine (UH-1H).

b. To determine engine characteristics throughout the flight envelope.

c. To investigate the helicopter's handling qualities in the expanded flight envelope resulting from the higher horsepower of the T53-L-13 engine.

d. To provide information for use in Army technical manuals and other service publications.

DESCRIPTION

3. The test helicopter was a prototype YUH-1D, S/N 60-6029, with an early production T53-L-13 engine installed. The T53-L-13 is a free turbine engine rated at 1400 shaft horsepower (shp). The UH-1H installation is derated to 1100 shp due to the helicopter power train limits. This engine is physically interchangeable with the T53-L-9 or T53-L-11 engine currently installed in the UH-1D helicopter. The increase in power and improved acceleration of the T53-L-13

engine is attained by modification of the axial compressor and the addition of variable inlet guide vanes, a second gas-producer turbine stage, and a second power turbine stage. This engine (as installed) weighs 530 pounds, an increase of 34 pounds over the T53-L-11. Five different engine air inlet configurations were tested. A detailed description of the helicopter is included in appendix IV.

Photo 1. YUH-1D/H Helicopter.

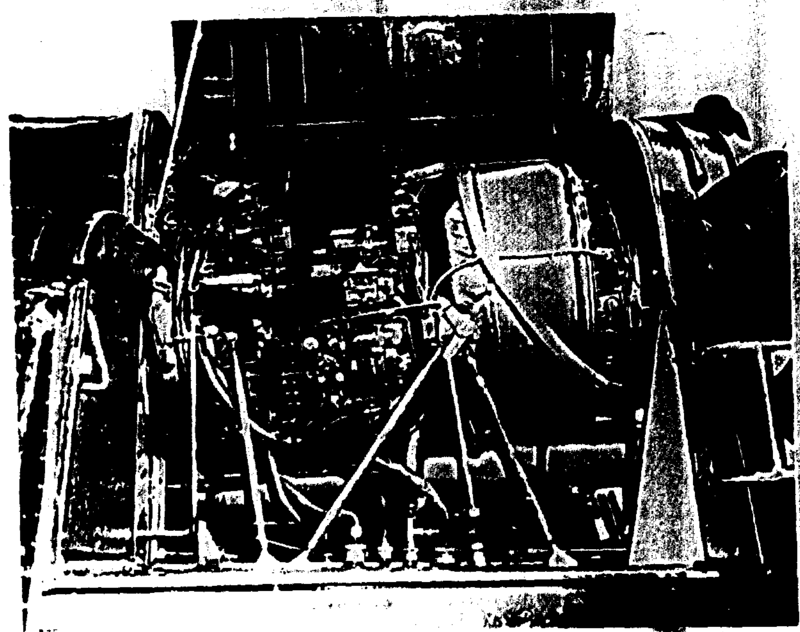

Photo 2. T53-L-13 Engine Installed in YUH-1H Left Side.

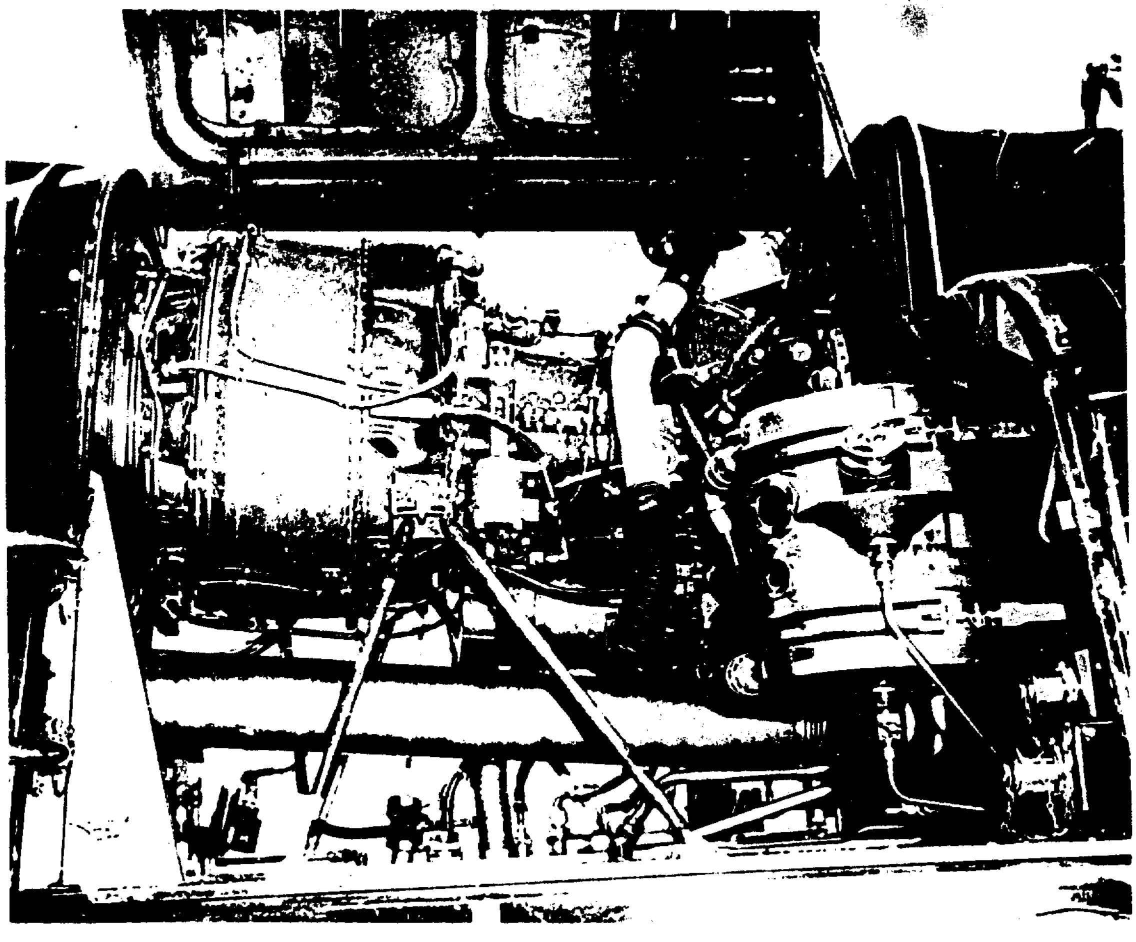

Photo 3. T53-L-13 Engine Installed in YUH-1H Right Side.

SCOPE OF TEST

4. The performance of the YUH-1H helicopter was quantitatively determined by hover, takeoff, climb and level flight tests. Handling qualities were qualitatively evaluated during each flight. The flight characteristics were investigated quantitatively in sideward and rearward flight, during approaches in winds, and while hovering in winds. Engine characteristics were investigated throughout the flight envelope.

METHOD OF TEST

5. Detailed test techniques and data analysis methods are included in appendix III.

CHRONOLOGY

6. The chronology of this test program is as follows:

Test directive issued	9 February 1966
Test aircraft received	19 November 1966
Tests started	20 December 1966
Tests completed	18 August 1967
Interim report published	August 1967
Draft report submitted	September 1969

RESULTS AND DISCUSSION

GENERAL

7. This section presents a detailed discussion of the test. The subjects covered are helicopter performance, helicopter stability and control, engine performance, engine operating characteristics, and other miscellaneous items. The performance of the test helicopter is significantly improved over that of the UH-1D helicopter, resulting in comparable performance at a 1000-pound heavier gross weight (grwt). The stability and control characteristics of the test helicopter were acceptable except for operations in the low-speed regime where control limitations were encountered. Engine performance and operating characteristics were good except for poor static droop and engine dynamic instability at topping power. These tests were conducted on a prototype UH-1H, and results may not be representative of production helicopters. Also, an adequate temperature range was not available to sufficiently determine the effects of compressibility over the operational temperature range. The results of this evaluation show that due to control limitations, vibration and possible structural considerations, the UH-1H helicopter in the configuration tested should not be operated at gross weights heavier than 9500 pounds.

PERFORMANCE

General

8. The most significant improvement of the UH-1H over the UH-1D is the increased hover performance provided by the additional power available from the T53-L-13 engine. For example, at a pressure altitude (H_P) of 6000 feet and 35°C (95°F), the UH-1D can hover out of ground effect (OGE) at 6740 pounds; the YUH-1H can hover OGE at 7400 pounds. At 9000 pounds on a standard day, the UH-1D can hover OGE at 3100 feet, and the YUH-1H can hover OGE at 8600 feet. An improvement in takeoff performance was also realized. Specific range performance has improved slightly due to lower specific fuel consumption of the T53-L-13 at cruise powers. A detailed investigation of the effects of compressibility showed that range will be decreased 20 percent or more when flying in cold ambient conditions due to higher power requirements. For example, at a 5000-foot altitude, an 8500-pound grwt, and recommended cruise speed, the specific range at +30°C is 0.212 nautical air miles per pound of fuel (NAMPP); and the specific range at -25°C is 0.164 NAMPP. Endurance of the YUH-1H is less than that of the UH-1D because of higher specific fuel consumption of the T53-L-13 at endurance power settings. The service ceiling of the YUH-1H is approximately 3000 feet higher than that of the UH-1D.

Hover Performance

9. Hover performance tests were conducted to determine the power required to hover under various conditions. Both the tethered and free-flight test methods were used. The tests were conducted over the following range of conditions: weight (thrust), 5500 to 9500 pounds; density altitude (H_D), 2800 to 12,000 feet; ambient temperature, -16° to +20°C; rotor speed, 314 and 324 rpm; rotor tip Mach number, 0.690 to 0.755; skid heights, 2- and 5-foot in ground effect (IGE) and 60-foot OGE; winds, less than 2 knots. Test data were reduced to nondimensional form and are presented in figures 7 through 9, appendix II. The fairings of these data are combined with the hover performance data of the UH-1D (ref 5, app I) and are presented in a nondimensional summary in figure 6, appendix II. Figures 4 and 5 show the power required to hover OGE and at a 2-foot skid-height IGE. The OGE hover performance and the 2- and 5-foot IGE hover performance at military power are shown graphically in figures 1, 2 and 3.

Photo 4. Tethered Hover.

Comparison of YUH-1H With UH-1D:

10. On a standard day above critical altitude, the YUH-1H can hover OGE at a 1000- to 1200-pound heavier grwt than the UH-1D, depending

upon altitude. Below the critical altitude where power is restricted by the UH-1H transmission torque limit, the difference in hovering capability decreases. Figure A shows the difference between the YUH-1H and UH-1D OGE power-limited hover performance. At a 9500-pound grwt (structural limit), the YUH-1H can hover OGE at 4000 feet on a standard day. On a 35°C day, the YUH-1H can hover OGE at a 600- to 850-pound heavier grwt than the UH-1D, depending on altitude. At sea-level (SL) pressure altitude, the YUH-1H can hover OGE at 9310 pounds. IGE hovering performance is improved similarly. These hover performance data are based on engine power available at the maximum allowable power setting (MP 30-minute limit). At most conditions, this performance cannot be safely achieved in adverse winds due to the insufficient directional control (para 55). For example, at the high-altitude site (11,700 feet), maximum thrust was limited by full left directional control requirement not by engine power available. Also, a reduction in rotor speed always affected directional control characteristics adversely and, for most conditions, degraded hover performance. All hover operations in the UH-1H should be conducted at 324 rotor rpm. Figures B, C and D show hovering performance ceilings as limited by both engine power and directional control. The hover capability as limited by directional control is affected by the center of gravity (cg) of the aircraft (para 50). Figure E shows the effect of rotor speed on power available and power required to hover at a representative condition.

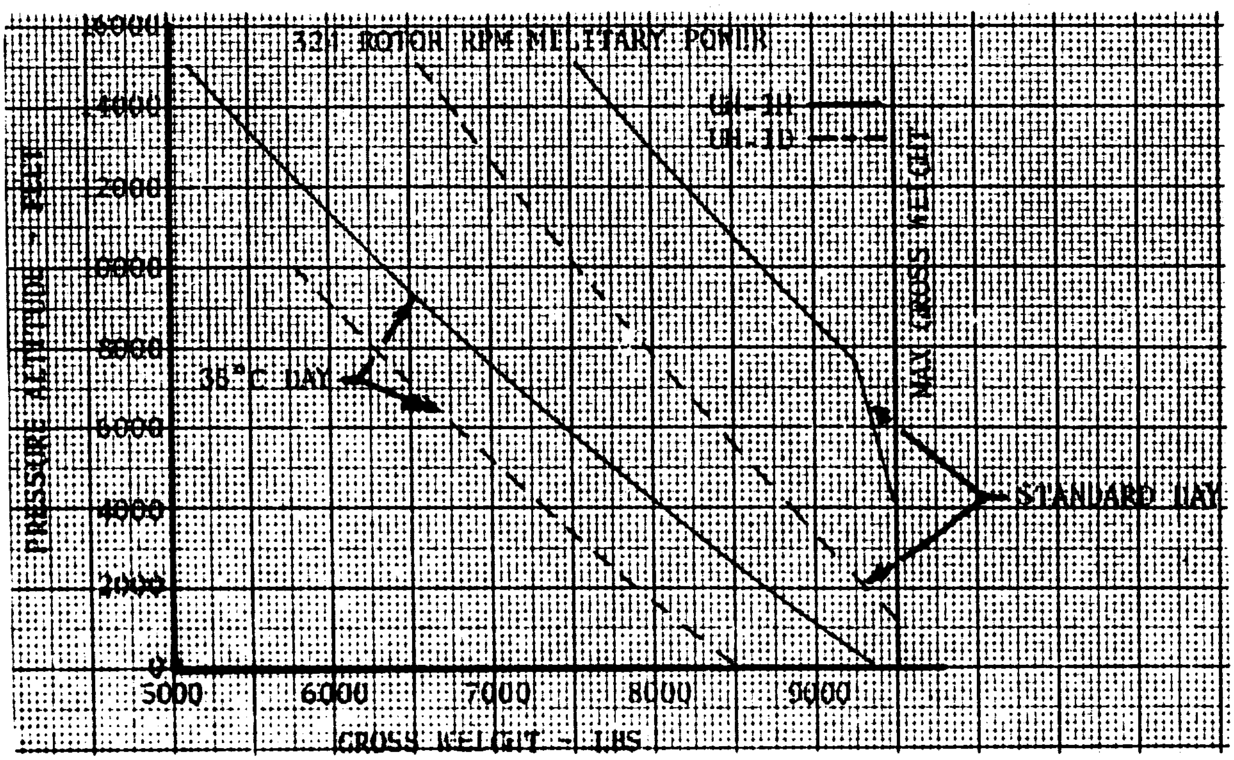

Figure A. Comparison of UH-1H and UH-1D OGE Hovering Ceilings.

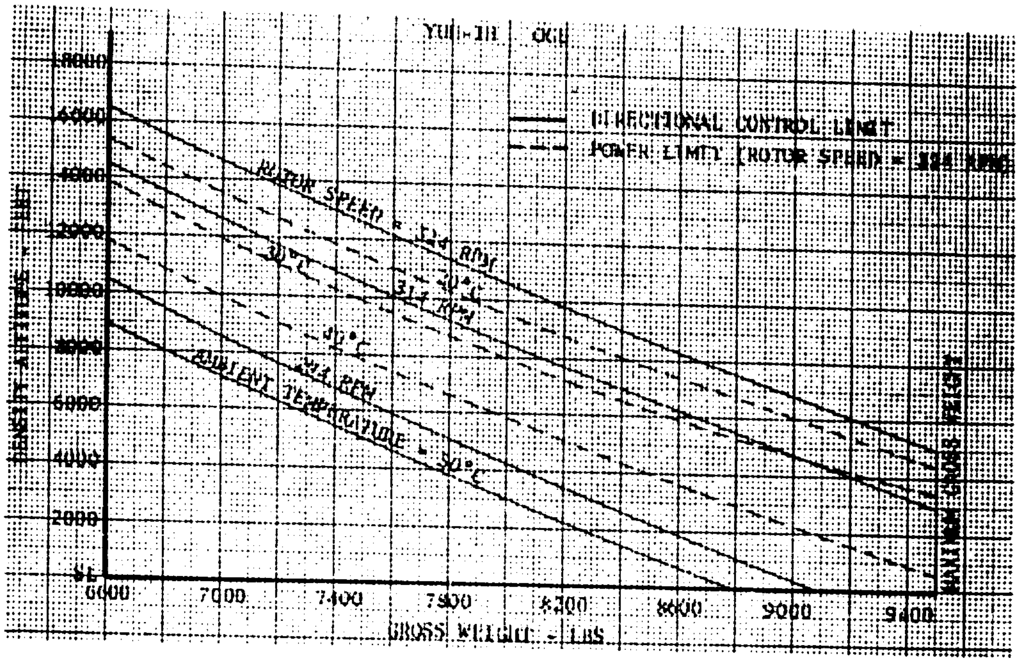

Figure B. Composite Power and Directional Control Limited Hovering Performance.

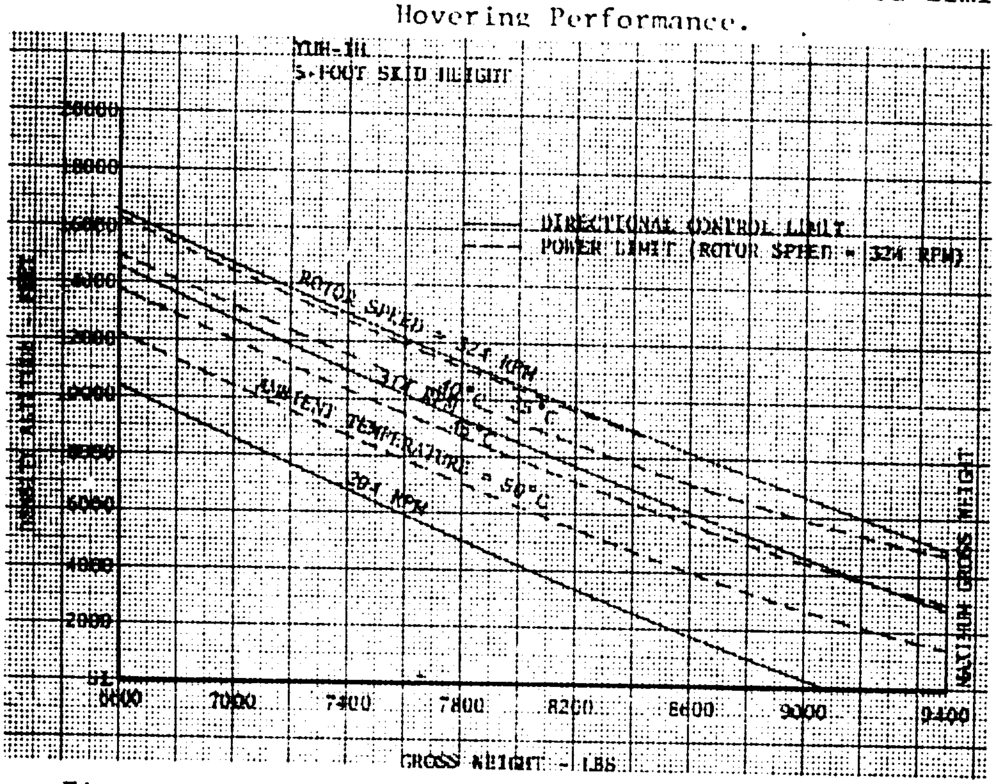

Figure C. Composite Power and Directional Control Limited Hovering Performance.

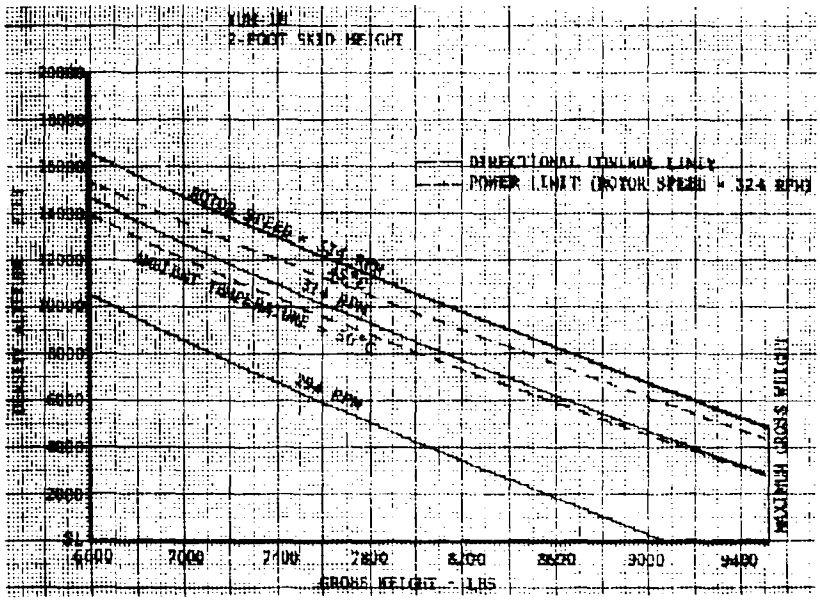

Figure D. Composite Power and Directional Control Limited Hovering Performance.

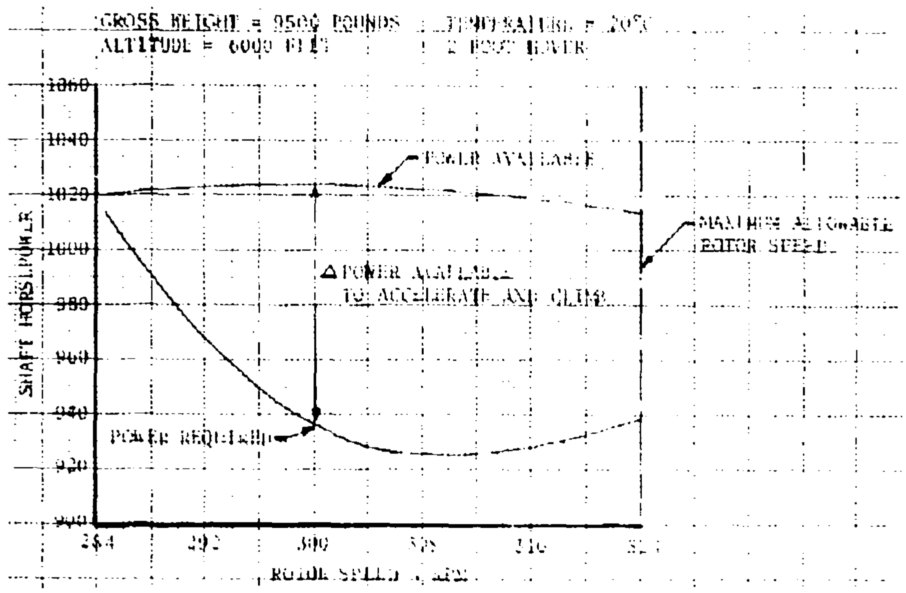

Figure E. Effect of Rotor Speed on Power.

Data Scatter:

11. A relatively large amount of scatter (0.00002 to 0.00003 power coefficient (Cp)) is present in the hover data (figs. 7 through 9, app II). This scatter is attributed to secondary aerodynamic effects of both the main and tail rotors. An analysis of the level-flight data coupled with a discontinuity in hover data at high thrust coefficient (C_T) (0.0044) indicates that main rotor compressibility effects and tail rotor stall effects are both present. Since both stall and compressibility are a function of the same parameters (angle of attack, temperature, tip speed, etc.), it is difficult to separate and analyze their effects. Insufficient data are available to analyze and summarize these "secondary effects" on hover performance. It is recommended that hover tests be conducted over a wider ambient temperature range with tail rotor power instrumentation so these secondary effects can be defined and hover data more accurately determined. The IGE hover data fairings (figs. 7 and 8) used in this report disagree with the hover data points. They are based on the fairings from reference 5, appendix I, extended to cover the Cp - C_T capabilities of the UH-1H. The difference decreases with increasing skid height, and the data agree OGE. This indicates that the discrepancy is caused by a difference in determining skid height. Rotor height is the significant parameter in determining ground effect. The difference in rotor to skid distance for the two sets of data is approximately 6 inches. This program was directed to make maximum use of reference 5 (UH-1D) data. Therefore, the reference 5 fairings were used for the lower C_T's rather than to repeat all hover skid heights and other conditions used for the reference 5 tests. Additionally, these fairings are conservative to account for unknown compressibility effects at colder temperatures compared to the test data from this program.

Takeoff Performance

12. Takeoff tests were conducted to determine the distance required to clear a 50-foot obstacle at various conditions. All data were obtained using the level acceleration technique. Other techniques were qualitatively investigated. The data were obtained in surface winds of less than 3 knots. The tests were conducted throughout the following range of conditions: gross weight, 7390 to 8930 pounds; density altitude, 5900 to 14,030 feet; ambient temperatures, 2° to 27°C; rotor speed, 324 rpm; cg location, mid. These conditions resulted in excess power coefficients (ΔC_p) from 0.08 x 10^{-5} to 11.47 x 10^{-5}. The ΔC_p method of data reduction was used. Test data are presented in figures 17 through 22, appendix II, and are summarized in figure 16. Figures 10 and 11 present comparisons of the UH-1D and YUH-1H dimensional takeoff performance for a standard day and for a 35°C day. Takeoff performance is shown in figures 12

through 15. The best takeoff performance is obtained at climb airspeeds below 35 knots calibrated airspeed (KCAS) (30 knots indicated airspeed (KIAS)); however, 30 KIAS is recommended for the following reasons:

a. This airspeed will minimize the time in the "avoid" area of the height-velocity diagram.

b. Near the 2-foot hover ceiling, optimum climbout airspeed (minimum distance airspeed) approaches this airspeed.

c. This airspeed is the minimum at which the airspeed system is reliable. Below this, airspeed must be estimated.

d. There is good performance with no possibility of "fall through" (insufficient power to climb) at this airspeed.

Takeoff Techniques:

13. Before these takeoff tests were conducted, the results of the UH-1D takeoff tests (ref 5, app I) were reviewed. This review showed that the "rpm bleed" technique gave the best performance at conditions where the helicopter could not hover OGE. The "rpm bleed" technique was not used because attempting to "bleed" rotor speed by increasing collective below critical altitude will result in UH-1H drive train overtorque and, under some conditions, will result in loss of directional control (para 55). The "level acceleration from a 2-foot hover" technique was used because it results in better performance than the normal "simultaneous climb and acceleration" technique at conditions where the helicopter cannot hover OGE. These same characteristics were confirmed for the UH-1H.

14. The "level acceleration from a 2-foot hover" takeoff technique was performed in the following manner: The helicopter was stabilized at a 2-foot skid-height hover with rotor speed set at 324 rpm. Military power was applied (limit torque or topping power, whichever was lower) by increasing collective pitch. The helicopter was then transitioned into forward flight, and a constant skid-height acceleration was made. Approximately 5 knots before desired climb airspeed was reached, rotation to climb attitude was initiated. The climb was then accomplished at constant airspeed and rotor rpm. As collective pitch was increased to demand power, the power turbine governor speed selector switch "beep" had to be increased to maintain the rotor speed at 324 rpm. When topping power could be reached without exceeding the torque limit, full "beep" was applied while applying collective. Rotor speed was maintained by adjusting the collective position. At conditions where takeoff capability was marginal (hovering capability between 2 and 5 feet), it was

necessary to level the helicopter just prior to obtaining translational lift to avoid ground contact. With less than a 2-foot hovering capability ground contact cannot be avoided. Ground contact will increase takeoff distance from 200 feet (minimum) to as much as 500 feet depending on severity and duration. When ground contact is inevitable, the aircraft should be leveled to decrease the severity and duration of the contact. After translational lift has been reached, it is necessary to apply forward stick pressure to maintain constant skid height. For all takeoff conditions tested, the higher power resulted in more left pedal required for the YUH-1H than for the UH-1D. At some conditions, full left pedal was necessary during takeoff.

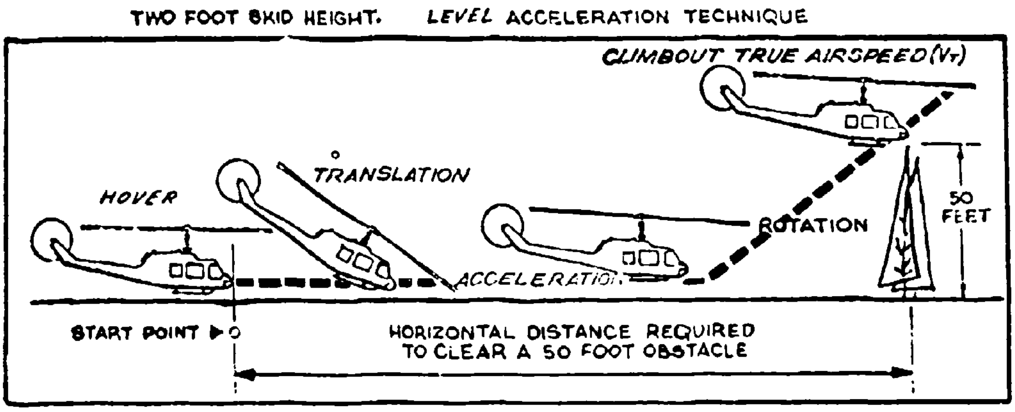

Figure F. Takeoff Flight Path.

Power Management During Takeoff:

15. The poor static droop characteristics of the YUH-1H greatly increased the pilot workload during takeoff because of the necessity to constantly monitor rotor speed. Over a wide range of ambient temperatures and the operating altitudes, the T53-L-13 engine is capable of exceeding the design torque limit of the helicopter power-train system. For this reason, it is important for the pilot to monitor the torquemeter as the primary power instrument. Copilot assistance in monitoring torque can be helpful in reducing the pilot workload. A more readable instrument incorporating an overtorque warning light would tend to prevent the pilot from exceeding the airframe limits. The incorporation of an automatic torque limiter with a manual override feature would reduce pilot workload and minimize aircraft component damage. The override feature would allow the pilot to exceed the torque limit on a one-time basis in an operational emergency.

Comparison of YUH-1H and UH-1D Takeoff Performance:

16. Figures G and H compare the YUH-1H and UH-1D dimensional takeoff performance for a standard day and a 35°C day. At a 9500-pound grwt on a standard day, the YUH-1H has approximately the same takeoff performance at a 5000-foot higher altitude than the UH-1D. At an 8500-pound grwt on a 35°C day, the YUH-1H takeoff performance is equivalent to that of the UH-1D at an approximate 4000-foot higher altitude. Nondimensional takeoff performance data of the UH-1D and YUH-1D are compared in figures I and J. The nondimensional method of data analysis was used for these takeoff tests. This method assumes that the average power required for unaccelerated flight during the takeoff is equivalent to the Cp required to hover at a reference skid height (2 feet). Any excess power is available for acceleration and climb. The effects due to change in the C_T are assumed to be accounted for in the change in power required to hover at the reference skid height. As was done in these tests, takeoffs are usually made at several different altitudes to verify this assumption. However, the large change in power available between the YUH-1H and UH-1D requires a relatively large difference in C_T for both aircraft models to operate at the same ΔCp. In dimensional terms, the YUH-1H must be at a significantly higher weight or altitude to have the same excess power as the UH-1D. At this higher weight or altitude, the same excess power results in decreased acceleration and climb (takeoff) performance as is shown in figures I and J. Therefore, nondimensional UH-1D takeoff performance cannot be used to derive UH-1H takeoff performance.

17. The hover capability is a good indication of takeoff performance as shown in figures 12 through 15, appendix II. Using the level-acceleration technique, a takeoff will require approximately 550 feet to clear a 50-foot obstacle where the YUH-1H can just hover OGE. With a 5-foot hover capability, the YUH-1H will require approximately 750 feet to clear a 50-foot obstacle. At a 2-foot skid-height hover capability, approximately 1000 feet is required to clear a 50-foot obstacle provided ground contact does not occur. Ground contact will cause the takeoff distance to increase an additional 200 to 500 feet.

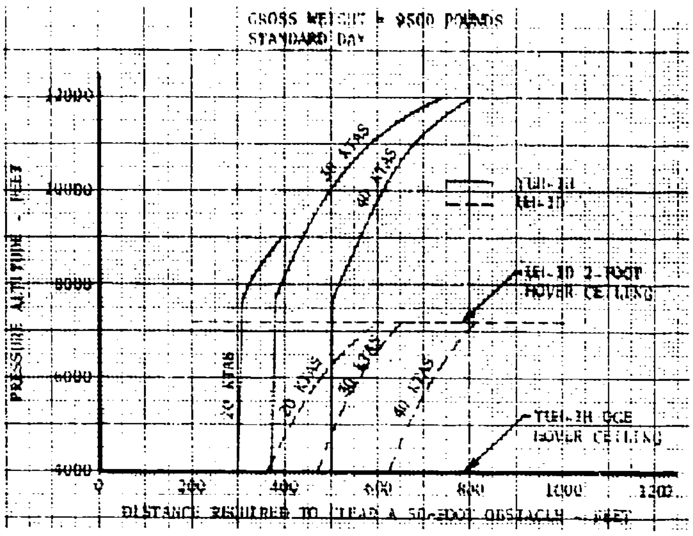

Figure G. Dimensional Takeoft Performance Comparison of YUH-1H and UH-1D.

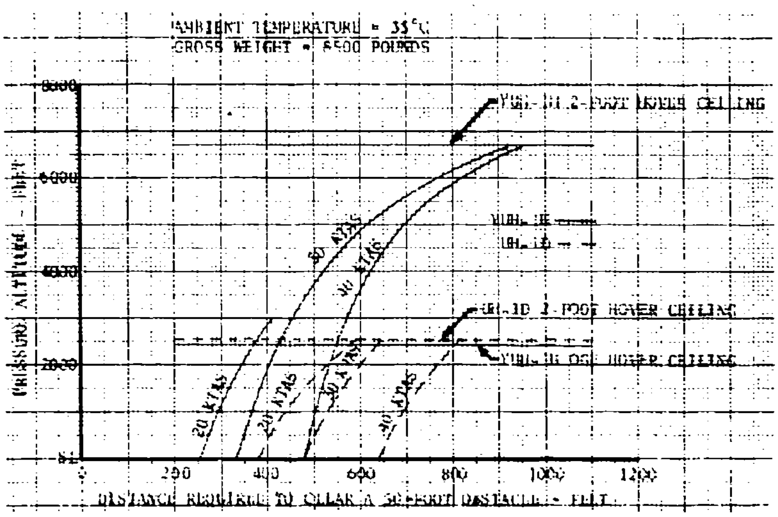

Figure H. Dimensional Takeoff Performance Comparison of YUH-1H and UH-1D.

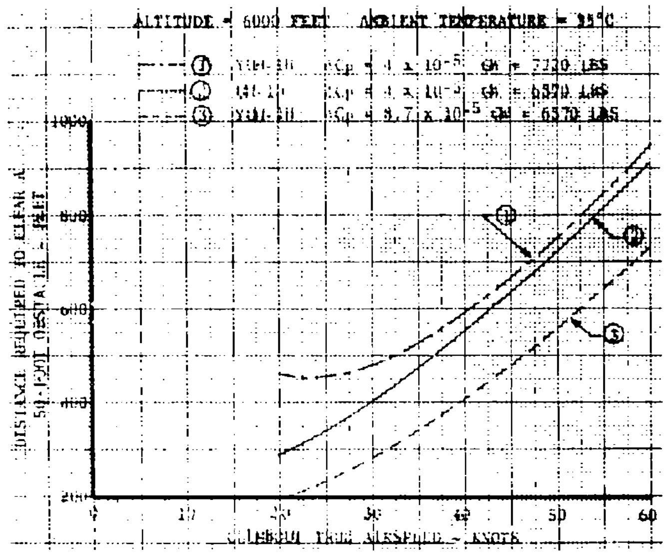

Figure I. Takeoff Performance Comparison of YUH-1H and UH-1D.

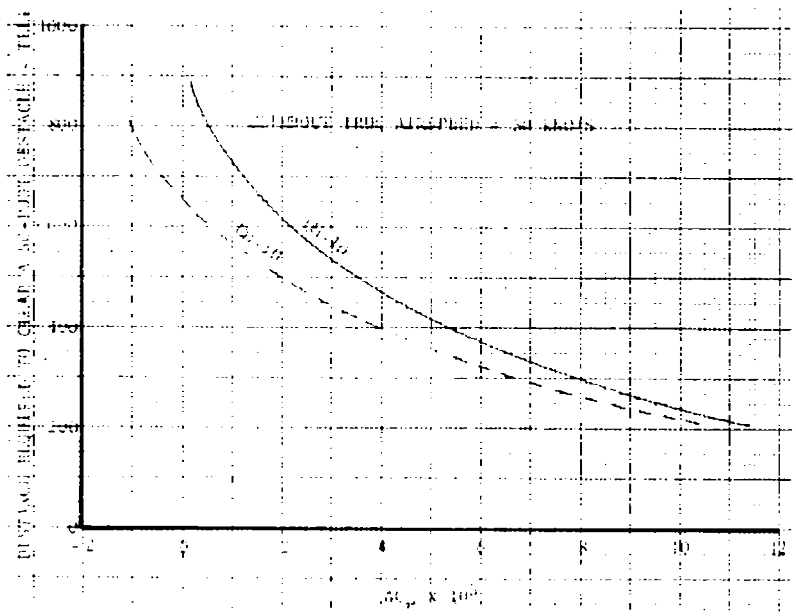

Figure J. Nondimensional Takeoff Performance Comparison of YUH-1H and UH-1D.

Takeoff Limitations:

18. Normal takeoff weight should be limited to those conditions at which the UH-1H can hover at a 5-foot skid height (fig. 2, app II). Maximum takeoff weight (emergency) should be limited to those conditions at which a 2-foot skid-height hover can be obtained (fig. 3). The criterion (loading the YUH-1H to the maximum weight at which it can just become airborne) often used for other UH-1's will result in severe overloading of the YUH-1H for many conditions. This criterion for determining takeoff weight will cause control problems and may result in structural damage.

Aborted Takeoff Performance

19. Low-speed decelerations were made to determine the distance required to abort a takeoff. The tests were conducted at the following conditions: average gross weights, 5800 and 9300 pounds; density altitude, 2200 feet; ambient temperature, 12°C; cg location, mid; surface wind, less than 2 knots; entry skid heights, 2 and 50 feet; entry airspeed, 10 to 49 KTAS. Distance required to decelerate and stop at a 2-foot hover is presented in figures K and L and also in figure 23, appendix II. A comparison of attitude and airspeed versus distance of one deceleration is shown in figure 24.

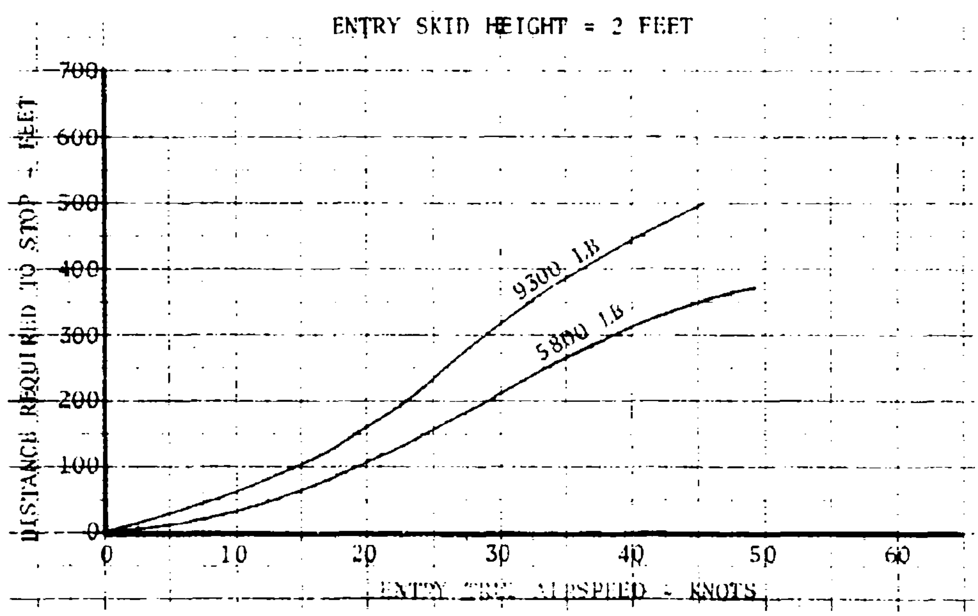

Figure K. Stopping Distance.

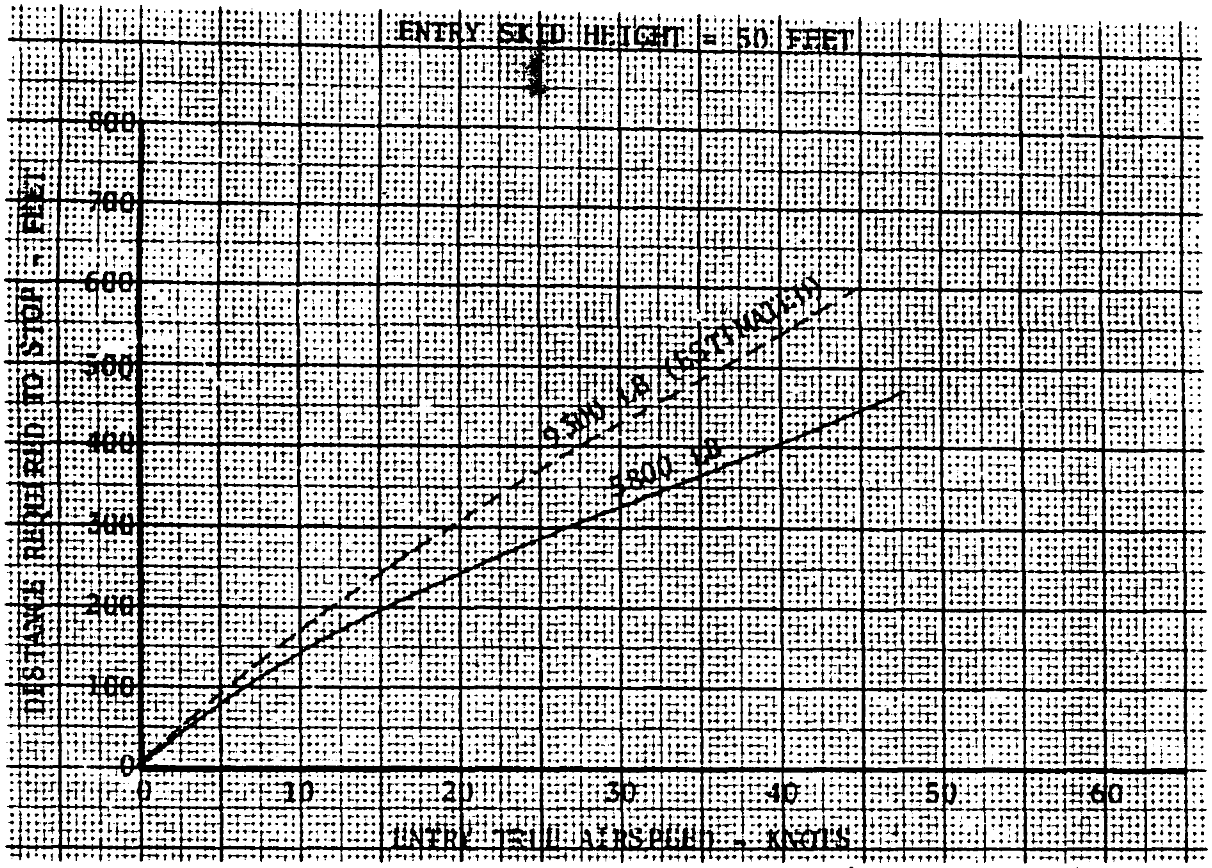

Figure L. Approximate Landing Distance.

Technique for Aborted Takeoff:

20. The pilot conducted a "normal deceleration" as if aborting a takeoff. Rotor speed was set at the minimum allowable steady state speed (314 rpm) prior to the deceleration, and no "beep" adjustment was made. The flare was executed so maximum steady state rotor speed (324 rpm) was not exceeded. Pitch attitude change did not exceed 20 degrees during any deceleration. Deceleration performance could be improved (stopping distance shortened) if a more extreme flare was used and rotor speed was allowed to go to the maximum power-on transient overspeed limit (331 rotor rpm/6750 engine rpm) or if "beep" adjustments were made during the deceleration. All entries were made from stabilized airspeeds using a ground pace vehicle as a speed reference. Flight path time histories were recorded using a Fairchild Flight Analyzer camera (photo 5). For the 50-foot entry height, altitude was decreased to a low skid height before forward speed was decreased. This information should be considered as a guide only. Since the data were obtained under one set of conditions, the effects of wind, runway slope, pilot technique and terrain were not determined.

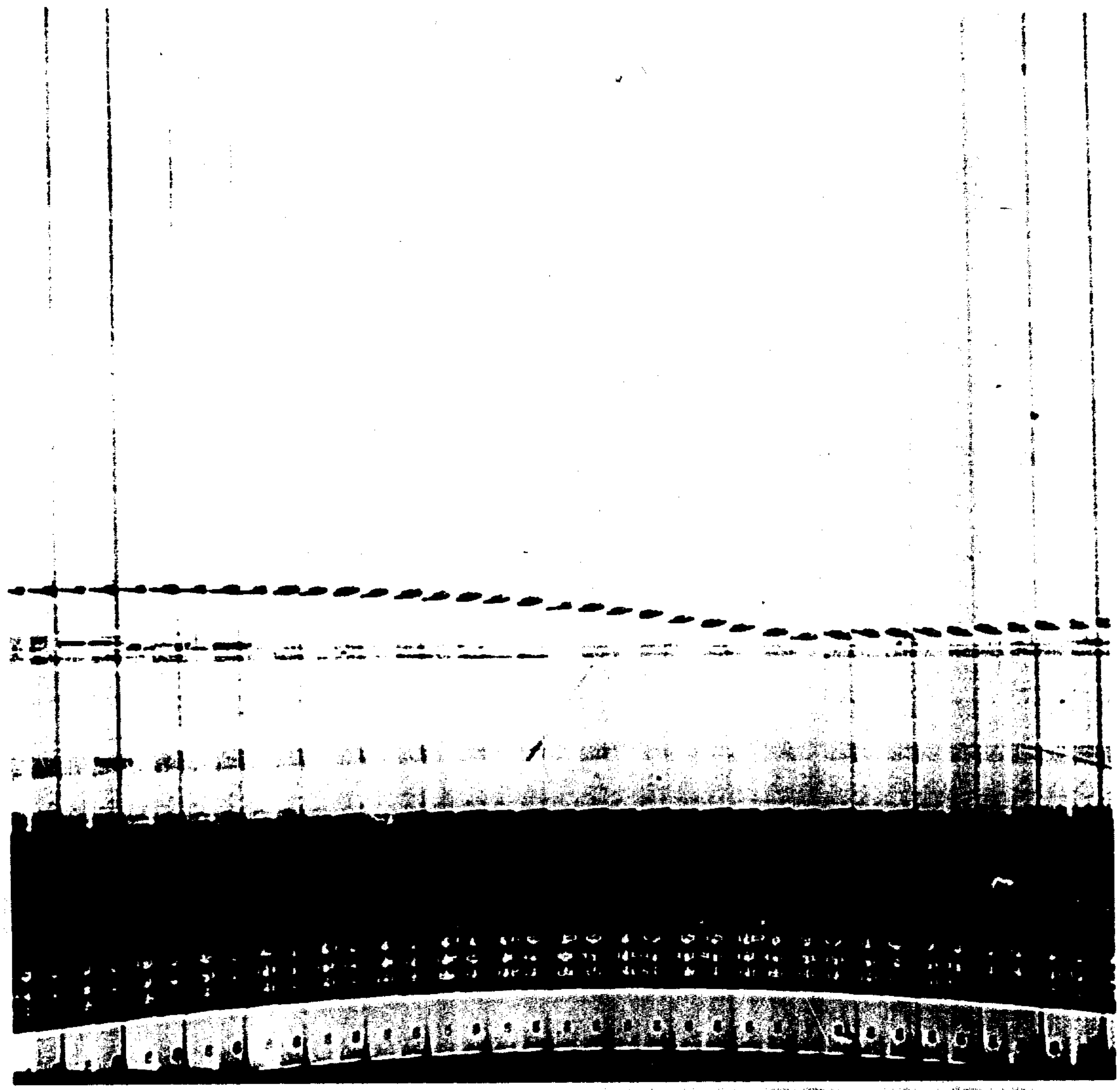

Photo 5. Deceleration From 50 Feet.

Landing Performance:

21. Although landing tests were not conducted, an insight into landing performance was acquired from the deceleration tests. Figure L shows the distance required to stop versus airspeed with an entry height of 50 feet at a 5800-pound average grwt. Distance is extrapolated for a 9300-pound grwt. This information can be used as a guide for normal landing distance over a 50-foot obstacle. During this test program, it was determined that a 5-foot hover performance capability was necessary to prevent hard landings when landing downwind in winds as light as 5 knots. Vertical or steep approaches (greater than 15 degrees) should not be made unless OGE hover capability is available.

Continuous Climb Performance

22. Continuous climb tests were conducted to determine climb performance and provide data for comparison of the YUH-1H to the UH-1D. The continuous climb test results were also used to verify the climb performance summary that was derived from the level-flight data. The continuous climbs were made at the following conditions: SL climb-start grwt, 7000, 8350, 8600 and 9500 pounds; density altitude, approximately 3000 feet to service ceiling; rotor speed, 324 rpm; cg location, mid; power, military and maximum continuous; airspeed, optimum climb and a constant 80 KCAS. The climbs were conducted in stable atmospheric conditions and corrected to standard-day conditions. Test results are presented in figures 31 through 34, appendix II. Rate of climb (R/C), time to climb, fuel used and distance traveled while climbing are presented as functions of weight and altitude in figures 27 through 30. They are presented for standard-day conditions at MP and optimum airspeed, and for the standard-day conditions at maximum continuous power and 80 KCAS. A maximum service ceiling at MP is presented in figure 25 which shows service ceiling (altitude at which R/C is 100 feet per minute (fpm)) as a function of weight and temperature. At the service ceiling, little maneuvering or change of airspeed can be made without causing the helicopter to descend. A 500-fpm climb capability (combat ceiling) allows a more reasonable margin for varying airspeed and maneuvering. This capability will result in an altitude decrease of approximately 1000 feet at optimum airspeed (fig. 26). A summary of test results is shown in table 1.

Table 1. Comparison of YUH-1H and UH-1D Climb Performance.

(standard day, military power, mid cg)

Start Weight (lb)	Sea-Level Rate of Climb			10,000-Foot Rate of Climb			Service Ceiling		
	UH-1D[1] (fpm)	YUH-1H (fpm)	Change (fpm)	UH-1D[1] (fpm)	YUH-1H (fpm)	Change (fpm)	UH-1D[1] (fpm)	YUH-1H (fpm)	Change (fpm)
9,500	1,540	1,580	+40	600	1,300	+700	12,500	15,200	+2,700
8,600	1,760	1,950	+190	980	1,680	+700	14,550	17,600	+3,050
7,000	2,490	2,700	+210	1,740	2,560	+820	19,700	24,000	+4,300
6,300	2,750	[2]3,200	[2]+450	2,070	[2]2,950	+880	23,300	[2]26,900	[2]+3,600

[1]UH-1D data obtained from reference 5, appendix I.
[2]Estimated.

Note: A rotor speed of 314 rpm for the UH-1D and 324 rpm for the YUH-1H was used to obtain the above data.

23. Maximum climb performance was investigated so that comparisons could be made with the UH-1D data. Best climb speed for the YUH-1H varies from approximately 40 KCAS at light weight and/or high altitude to 60 KCAS at heavy weight and low altitude. At high R/C at best airspeeds, the YUH-1H exhibited unstable characteristics, and the desired airspeed was difficult to maintain. It was not unusual to encounter airspeed variations of 2 to 5 knots even though much of the pilot's attention was devoted to maintaining airspeed. A climb speed of 80 KCAS and maximum continuous power setting are recommended for the following reasons: Generally, the YUH-1H has satisfactory climb performance; flying qualities are significantly improved; and collective setting (power) is lower; therefore, if power failure occurs, autorotational entry characteristics will be better. A maximum performance climb and a "normal" climb at maximum continuous power and 80 KCAS are compared in figure M.

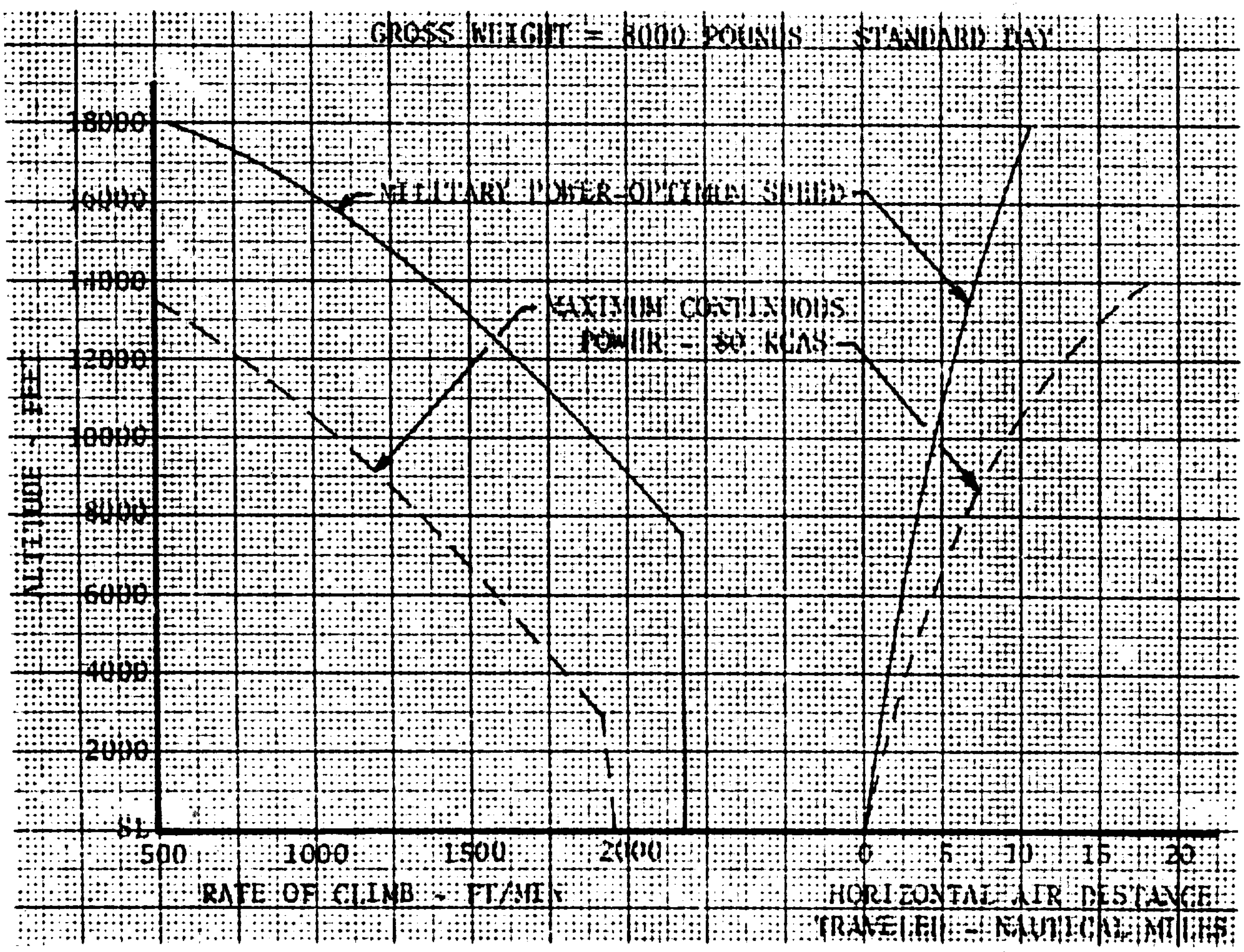

Figure M. Climb Performance Comparison.

Sawtooth Climb Performance

24. Sawtooth climb tests were conducted to determine the effects of airspeed, rotor speed, power and weight on climb performance. The data were obtained during stabilized climb segments in stable atmospheric conditions over the following range of conditions: gross weight, 6500 to 9500 pounds; density altitude, 5000 feet to service ceiling; power, minimum required for level flight to maximum test engine power available; rotor speed, 314, 319 and 324 rpm; cg location, mid. Results of these tests are shown in figures 35 through 37, appendix II.

Climb Airspeed Schedule:

25. The maximum R/C airspeed schedule was determined both from level-flight data at minimum power required and from sawtooth climb tests. True airspeed for maximum R/C was determined to be uniquely a function of C_T. Rotor speed, gross weight, altitude and ambient temperature (Mach number) affect optimum climb speed only as they affect C_T. Optimum climb speed coincides with minimum power required speed up to a C_T of 0.0038. Above this C_T, optimum climb speed is as much as 8 KTAS below the minimum power required speed. The optimum climb speed schedule and an example of performance variation with speed are shown in figure 35, appendix II. All airspeeds of a climb series were flown at the same altitude and power and were corrected to the average weight (fig. 36).

Weight Effect:

26. The effect of gross weight on climb performance was determined by flying sawtooth climbs at various gross weights, altitudes and rotor speeds. All sawtooth climbs of a set were flown at the same altitude and power. The optimum climb speed schedule (para 25) was used for all weight variation climbs. Test results are presented in figure 36, appendix II. The climb correction factor for weight (K_W) was found to be an explicit function of C_T. K_W varies from approximately 0.50 at a C_T of 0.0020 to 3.15 at a C_T of 0.0052. It is interesting to note that the R/C variation due to weight is large enough at heavy weights and high altitudes that fuel burn-off produces a significant change in R/C. For example, at 9500 pounds and 15,000 feet, the R/C variation is approximately 1.25 ft/min/lb. Fuel burn-off rate is approximately 540 lb/hr or 9 lb/min. Therefore, the change in R/C due to fuel burn-off is approximately 11 ft/min^2.

Rotor Speed Effect:

27. Figure 36, appendix II, also shows the effect of rotor speed variation on R/C. At low altitudes (below 10,000 feet) and at

heavy gross weights (above 7500 pounds), maximum rotor speed (324 rpm) produces the best climb performance. At gross weights below 7500 pounds and at altitudes above 10,000 feet, 314 rpm gives slightly higher R/C. This may be due to the increased compressibility effects caused by colder temperatures at higher altitudes. At higher gross weights, the effects of blade stall apparently predominate over compressibility. Maximum rotor speed (324 rpm) was used for the continuous climbs (para 22) at all weights and altitudes for the following reasons: It gives the best performance for most conditions; vibration is less at the maximum rotor speed; slightly higher power can be used at the UH-1H torque limit (para 56); and autorotational entry characteristics would be better at the higher rotor speed and lower collective pitch setting.

Power Effects:

28. Sawtooth climb tests were conducted to determine the effect of power variation on R/C. All sawtooths of a series were flown at the same density altitude and corrected to the same gross weight using figure 36, appendix II. A rotor speed of 324 rpm (para 27) was used for all tests. Test results are shown in figure 37. The climb power correction factor (K_p) varied from 0.77 to 0.84. A slight trend of increasing K_p with increasing weight and altitude was evident. This trend was smaller than the random scatter. The scatter may be due to variations in temperature which would change the effects of compressibility and K_p. The tests were conducted at temperatures from 5° to 15°C above standard-day temperature. A constant K_p of 0.80 was used for all corrections. This compares with the UH-1D results shown in reference 5, appendix I, which indicates a K_p of 0.738. For all conditions tested, increased power produced higher R/C.

Level Flight Performance

29. Level-flight performance tests were conducted to determine power required for level flight, engine characteristics, airspeed limits, range and endurance. Tests were conducted over the following range of conditions: gross weight, 6990 to 9500 pounds; density altitude, 2860 to 20,970 feet; ambient temperature, -25° to +28°C; rotor speed; 314 and 324 rpm, cg location; mid, forward and aft; airspeed, 20 to 128 KTAS; configurations, bleed air heater ON and OFF, cargo doors open and closed. These conditions resulted in the following range of nondimensional values: advance ratio (μ), 0.06 to 0.26; C_T, 0.00348 to 0.005278; advancing blade tip Mach number (M_{AT}), 0.738 to 0.934; C_P, 0.000134 to 0.000382. Test results are shown in figures 68 through 82, appendix II. Nondimensional data, including data from reference 5, appendix I, are summarized in figures 66 and 67. Range and endurance data are summarized in figures 38 through 65.

Data Analysis:

30. The classical nondimensional method of analysis of C_P, C_T and μ was used. It was necessary to add the fourth variable, M_{AT}, to account for secondary aerodynamic effects (para 32). All test data were converted to these four parameters and are summarized in figures 66 and 67, appendix II. The data were analyzed empirically by iterative carpet and cross plotting. Other methods have been developed to simplify analysis by using some standardized compressibility condition and a correction for nonstandard compressibility (*ie*, ΔC_P versus ΔM [change in power coefficient versus change in tip Mach number], ref 8, app I). However, to use these methods several simplifying assumptions must be made. Since the purpose of these tests was to investigate the expanded portion of the flight envelope, only the upper half of the C_T range was investigated. UH-1D data (ref 5) were used to aid in extrapolation of the lower portion. Figure N shows the range of Mach numbers and advance ratios obtained during these tests and the total possible range that might be experienced operationally. The range of test conditions was less than half the operational range. To give some indications of the effects of compressibility, the test data were extrapolated to more extreme ambient conditions and also interpolated to conditions not actually flown. This was accomplished in the following manner: Mach numbers/C_P's were interpolated/extrapolated linearly to the desired conditions; the C_T/C_P was interpolated/extrapolated parabolically; the μ/C_P's were interpolated/extrapolated parabolically; and data were then converted to dimensional form. The results of this analysis are presented in the range and endurance summaries shown in figures 38 through 65. Where extrapolation was necessary, the data are thus labeled. It is recommended that tests be conducted at more extreme ambient conditions (particularly colder temperatures) to verify the predicted values.

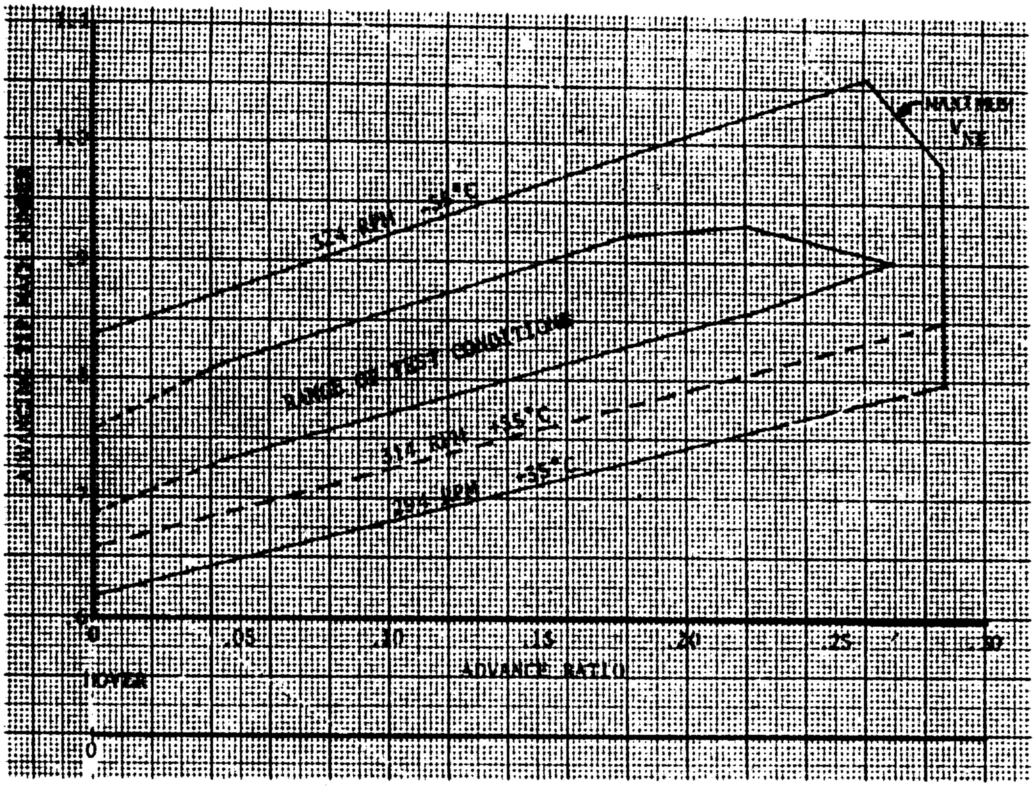

Figure N. Advance Ratio - Mach Number Range.

UH-1D Versus YUH-1H Drag:

31. Although prototype airframes were used for both the UH-1D test (ref 5, app I) and this test, an increase in drag was evident during these tests. The effect of this drag difference on power required is shown in figure O for one condition. This drag difference may be due to one or more of the following reasons:

a. At equivalent power output for a given condition, the T53-L-13 engine produces 20 to 40 pounds less net engine exhaust thrust than the T53-L-9 or the T53-L-11 engine (refs 9 and 10). Since thrust is not considered in the level-flight data analysis, it would show as drag increase.

b. The UH-1B tail boom used on the prototype airframes had a different synchronized elevator rigging procedure. Although both test vehicles used the UH-1D elevator, the test outlined in reference 5, appendix I, may have used the production UH-1D rigging procedure which is incorrect. This could cause a change in fuselage attitude and a difference in drag.

c. Various engine-air-inlet configurations were used for these tests; however, no significant difference in drag was found.

d. Some other external variance in the airframes may not have been apparent. In addition, there may be drag differences between the prototype and production airframes. In future phase D evaluations, current production model airframes should be used to preclude this problem.

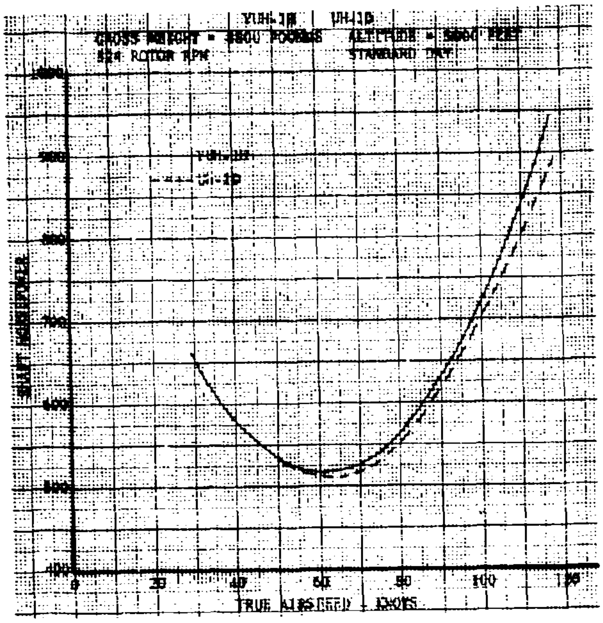

Figure O. Power Required Comparison.

Secondary Effects:

32. Secondary aerodynamic effects significantly influence the level flight performance of the YUH-1H. These "secondary effects" include both blade stall and compressibility effects. Blade stall is usually considered to be limited to the retreating side of the rotor disk. The drag rise due to compressibility is a function of the amount the actual Mach number is above the critical Mach number (Mach number where compressibility drag becomes significant) for the airfoil section. Critical Mach number, in turn, is a function of angle of attack. For the 0012 airfoil used on the UH-1H, the critical Mach number varies from approximately 0.3 at stall angle of attack to 0.72 at zero angle of attack. Therefore, the increase in power due to compressibility can occur both on the advancing side (low angle of attack, high Mach number) and on the retreating side (higher angle of attack and lower Mach number). The level-flight power-required data for the UH-1H correlated best with advancing rotor tip Mach number, indicating that advancing blade compressibility is

most significant. Figure P shows the effect of decreasing ambient temperature (increasing tip Mach number) on power required at one flight condition. Compressibility effects were present at all flight conditions encountered during these tests. Compressibility becomes most adverse at cold temperatures. Figure Q shows the effect of rotor speed on power required as a function of weight at one flight condition. The higher rotor speed (higher tip Mach number) requires less power at the higher weights which indicates that effects due to increased blade angle of attack dominate effects of increased Mach number. This can also be seen in the nondimensional summary (figs. 66 and 67, app II) as a sharp increase in C_P at constant Mach number and high C_T. The increase in power with increased blade angle of attack may be due to either retreating blade stall or decreased critical Mach number, or both.

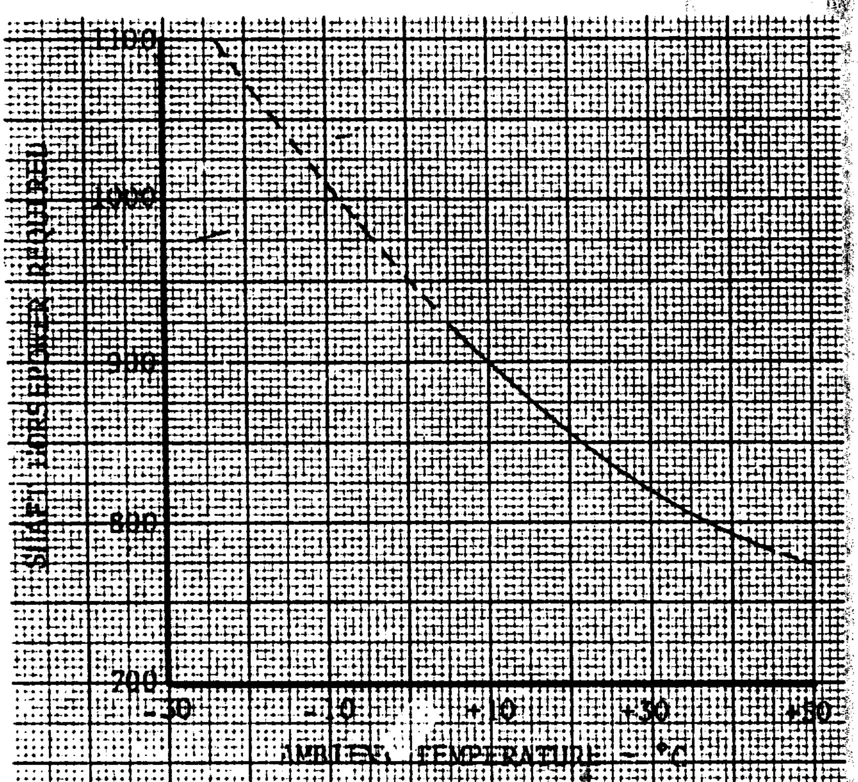

Figure P. Effect of Ambient Temperature on Power Required.

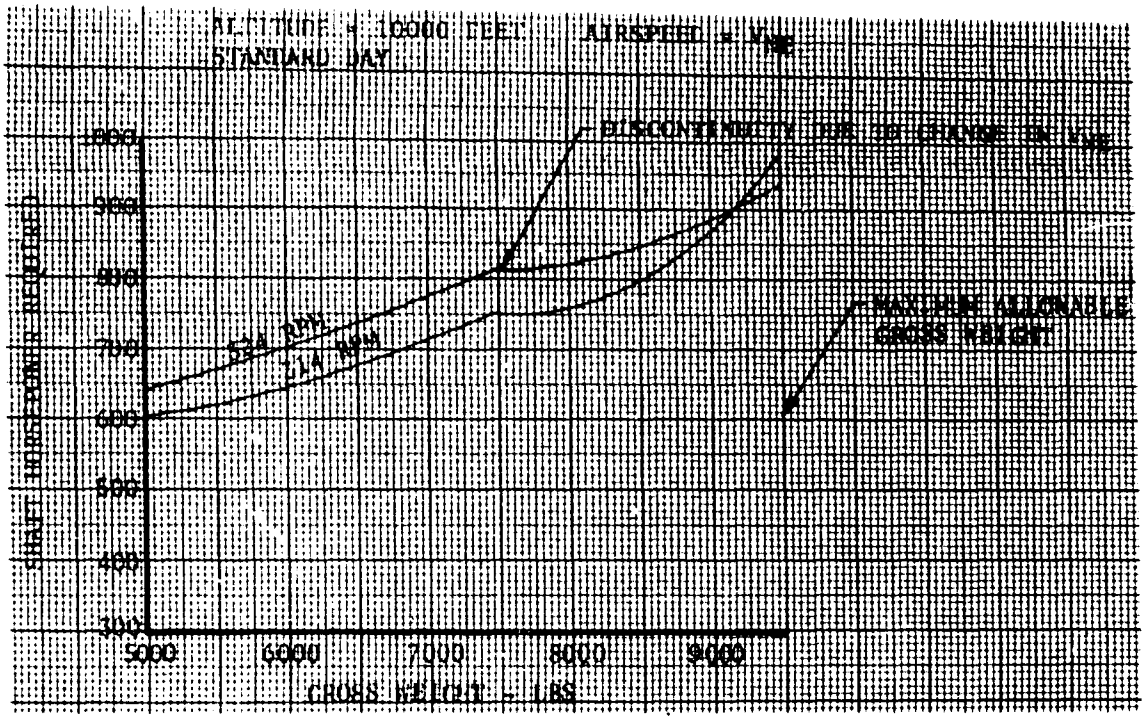

Figure 9. Effect of Rotor Speed and Weight on Power Required.

Other Effects:

33. The effects of aircraft longitudinal cg, cargo doors open, bleed air and sideslip on level flight performance were also investigated. A forward longitudinal cg slightly increased the power required (fig. 80, app II). Compared to mid cg, an aft cg (fig. 81) had no effect. With the cargo doors fully open, there was no significant effect with zero sideslip (fig. 82). With anti-icer ON and bleed air heat fully ON, range was reduced approximately 5 percent at the condition tested (fig. 71). The effect of sideslip on power required was investigated at a 9500-pound grwt and the 5000-foot H_D. Because of the poor effective dihedral and static lateral stability characteristic, relatively high angles were attained before sideslip became apparent to the pilot by either visual or kinesthetic cues. At cruise speed (108 KTAS), sideslip became apparent between 4 and 6 degrees. At minimum power speed (62 KTAS), sideslip was not apparent until the angle reached 6 to 8 degrees. These sideslip angles corresponded to a one-quarter to one-half ball width slip-ball displacement. Less than 15 additional horsepower were required at these conditions. Therefore, unnoticeable sideslip would have a very small effect on range performance. All other performance flight tests were conducted with zero sideslip and the cargo mirror removed. Reference 11, appendix I, shows that the addition of the cargo mirror will require up

to 55 additional horsepower for level flight at 120 KTAS and will significantly reduce the range performance. No level-flight performance tests were made with external stores. Range performance for a drag increase of 10 square feet in equivalent flat plate area (based on additional engine shp required) is shown in figure 41. This drag would approximate that of a twin XM23 gun or external fuel tank installation. The range performance should be determined with all normally carried external stores by additional testing.

Airspeed Limits:

34. Airspeed at various conditions were limited by velocity never exceed (V_{NE}) (structural), vibration or maximum power available. No stability and control problems were encountered which would limit maximum speed at the conditions tested. At colder ambient temperatures than those encountered during these tests, stability and control problems may be caused by increased compressibility effects. In general, maximum airspeed was limited by V_{NE} at standard or higher temperature below 10,000 feet. For lower temperatures, airspeed would be maximum power limited (UH-1H torque limit). Above 10,000 feet, airspeed was usually vibration or power limited. As altitude was increased, vibration became severe at 10 knots or more below V_{NE}. It appears that forward speed was limited to an advancing tip Mach number of approximately 0.92. Although this Mach number was slightly exceeded several times, severe vibration produced large two-per-revolution cyclic stick forces and displacement through the boost system and caused the flight instruments to become unreadable. The present V_{NE} of the UH-1H is 120 KCAS at a 7500-pound grwt or below from SL to 2000 feet with a 3-KCAS decrease for each increase of a 1000-foot H_D above 2000 feet, and a 1-KCAS decrease for each additional 200 pounds above 7500 pounds. In view of the vibration encountered during these tests, a more realistic V_{NE} would be as follows: 120 KCAS at 7500 pounds or below from SL to a 3000-foot H_D with a 4-KCAS decrease for each increase of 1000 feet H_D above 3000 feet and a 1-KCAS decrease for each additional 200 pounds above 7500 pounds.

35. Minimum airspeed was limited by power available or minimum usable indicated airspeed of the test boom system which was approximately 15 KIAS. Power recorded during the sideward and rearward flights gives some indication of power required for low-speed flight. Skid height was not accurately maintained; therefore, ground effect could not be determined, and the data could not be summarized. However, these data show that IGE power remains constant or increases slightly up to approximately 15 KTAS. Above this airspeed, a definite decrease was observed. This is also indicated by the settling tendencies of the UH-1H just prior to gaining translational lift during takeoff. Winds up to 15 KTAS will not improve IGE hovering performance and may have an adverse effect.

Range:

36. Range performance data are summarized in figures 39 through 41, appendix II, for a standard day, two rotor speeds and a drag increase which approximates external stores. Range data at different ambient temperatures are summarized in figures 42 through 47. Optimum cruise altitude data are summarized in figure 38. All of the range summaries were derived from the nondimensional power-required data (figs. 66 and 67), specification fuel flow corrected for installation losses and the particular conditions of each test. Five-percent conservatism was not applied. All of the engine test data showed that fuel flow was 2 to 4 percent below specification. However, fuel consumption should be investigated on high-time T53-L-13 engines to determine the variation due to engine use. Specific range points are shown together with a fairing derived from the specification fuel-flow data for a standard day (figs. 68 through 82).

Cruise Airspeed

37. It is recommended that the cruise airspeeds presented be limited to the lowest of the following conditions: structural limit airspeed (V_{NE}), airspeed for maximum continuous power setting, and maximum airspeed for the 99-percent maximum specific range (V_{99}). For most conditions, cruise speed was limited by V_{NE}. For heavy gross weights and high altitudes, cruise speed was limited by maximum continuous power. For light gross weights and low altitudes, cruise speed was determined by V_{99}. As ambient temperatures increased above standard day, the weight and altitude at which cruise speed was limited by maximum continuous power decreased due to decreasing power available. The decrease also occurred at temperatures below standard day because of increasing power required due to compressibility effects even though power available increased. Data extrapolated to extremely low temperatures (-30° to -50°C) indicate that at lighter gross weights, cruise speed for V_{99} drops off rapidly with decreasing weight to just slightly above minimum power speed because the advancing tip Mach number is near or above Mach 1. Since decreasing airspeed has only a slight effect on range, the range data have been presented only for V_{NE} at low temperatures. At these cruise speeds, vibration levels decrease as airspeed is reduced and reach a comfortable level at an approximate 80- to 90-knot cruise speed at low altitudes. Cruising at these reduced airspeeds decreases range performance. The additional advantage of the recommendations for a more realistic V_{NE} (para 34) is that V_{NE} would approach maximum performance cruise speed throughout the UH-1H envelope. Cruise information (fuel flow, true airspeed and power) could be presented along with V_{NE}.

UH-1D and UH-1H Comparisons

38. Figure R shows a comparison of UH-1D and YUH-1H range performance. Both the specific range and the best cruise airspeeds of the YUH-1H represent significant improvements over those of the UH-1D even though the effective drag (para 31) of the UH-1H has been increased. The reason for these improvements is the lower specific fuel consumption of the T53-L-13 engine as compared to the T53-L-9 and the T53-L-11 engines at cruise power. Figure 123, appendix II, shows that the specific fuel consumption of the T53-L-13 crosses below the specific fuel consumption of the T53-L-9 at approximately 700 referred shp. This improved range performance may not be apppar-ent to the user since the YUH-1H can be operated at an approximate 1000-pound heavier grwt than the UH-1D at the same ambient conditions.

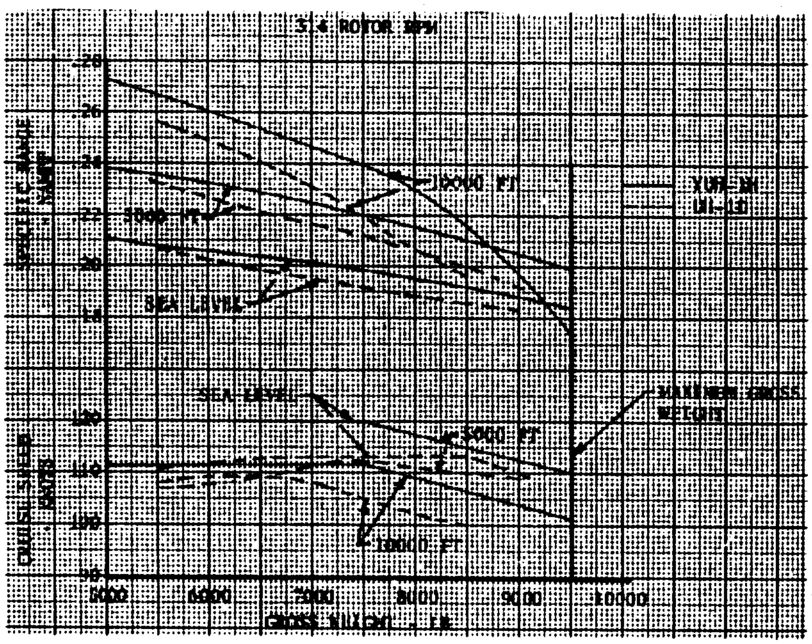

Figure R. Range Comparison YUH-1H and UH-1D.

Temperature Effect

39. The effect of ambient temperature on range performance at one condition is shown in figure S. There are two opposing reasons why temperature affects range performance which are as follows: At a constant power, fuel flow increases slightly with increasing temperature; and as temperature decreases, power required for a constant airspeed increases due to the effects of compressibility. At most temperatures, the effects of compressibility dominate, and range

performance (specific range) decreases significantly with decreasing temperature. In the example shown in figure S, range decreases approximately 30 percent when temperature decreases from +30° to -50°C. Part of this effect is due to the extraction of bleed air from the engine. At temperatures below 0°C, range data are presented for bleed air ON since the heat and/or anti-icing systems would normally be used at these conditions. For hot temperatures (above 30°C), range decreases with increasing temperatures. At these temperatures, compressibility effects become small, and the change in fuel consumption with temperature predominates. In addition, as temperature increases, power available decreases and may limit airspeed below best cruise airspeed.

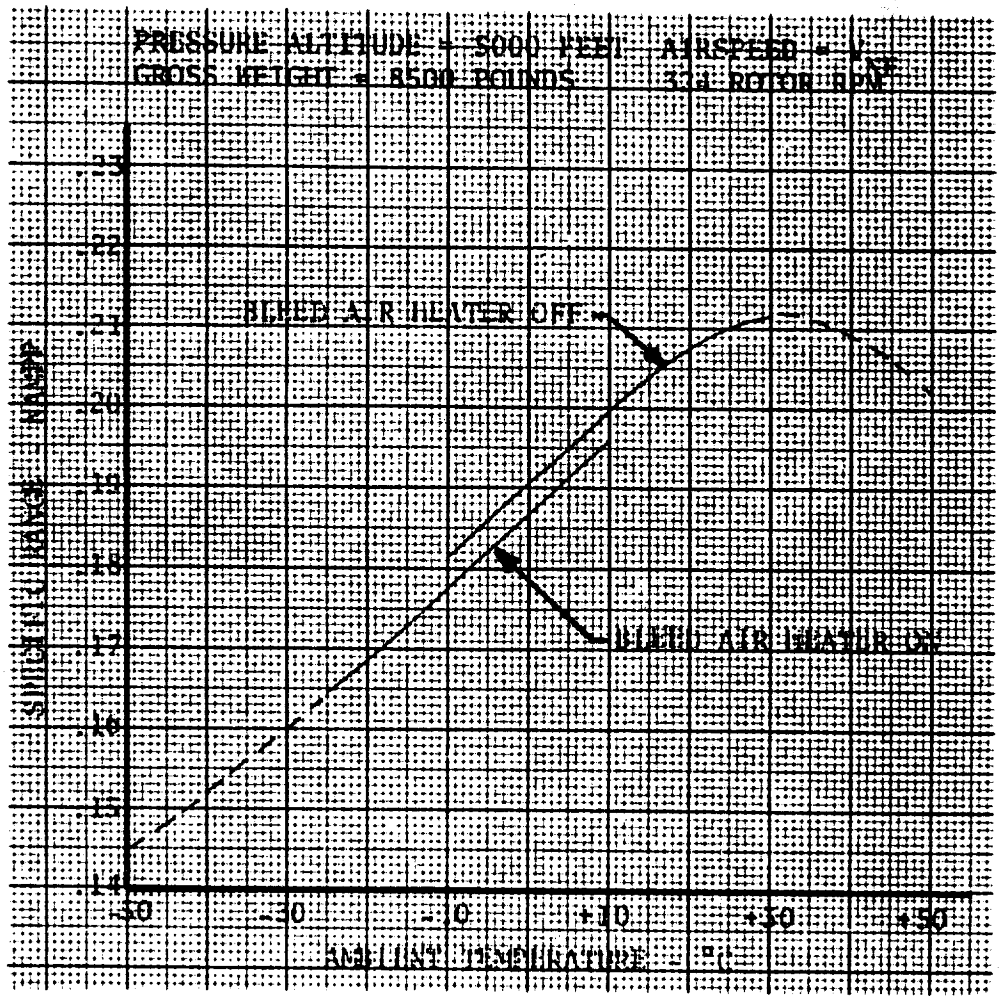

Figure S. Temperature Effect on Specific Range.

Rotor Speed

40. Figure T shows the effect of rotor speed on specific range at one altitude and temperature condition. Except at heavy weights and high altitudes, minimum allowable power-on rotor speed (294 or 314 rpm) produces the best range performance. However, vibration may preclude the use of minimum rotor speed at best cruise airspeed. For this reason, most of the range performance data are presented for 324 rpm (maximum steady state).

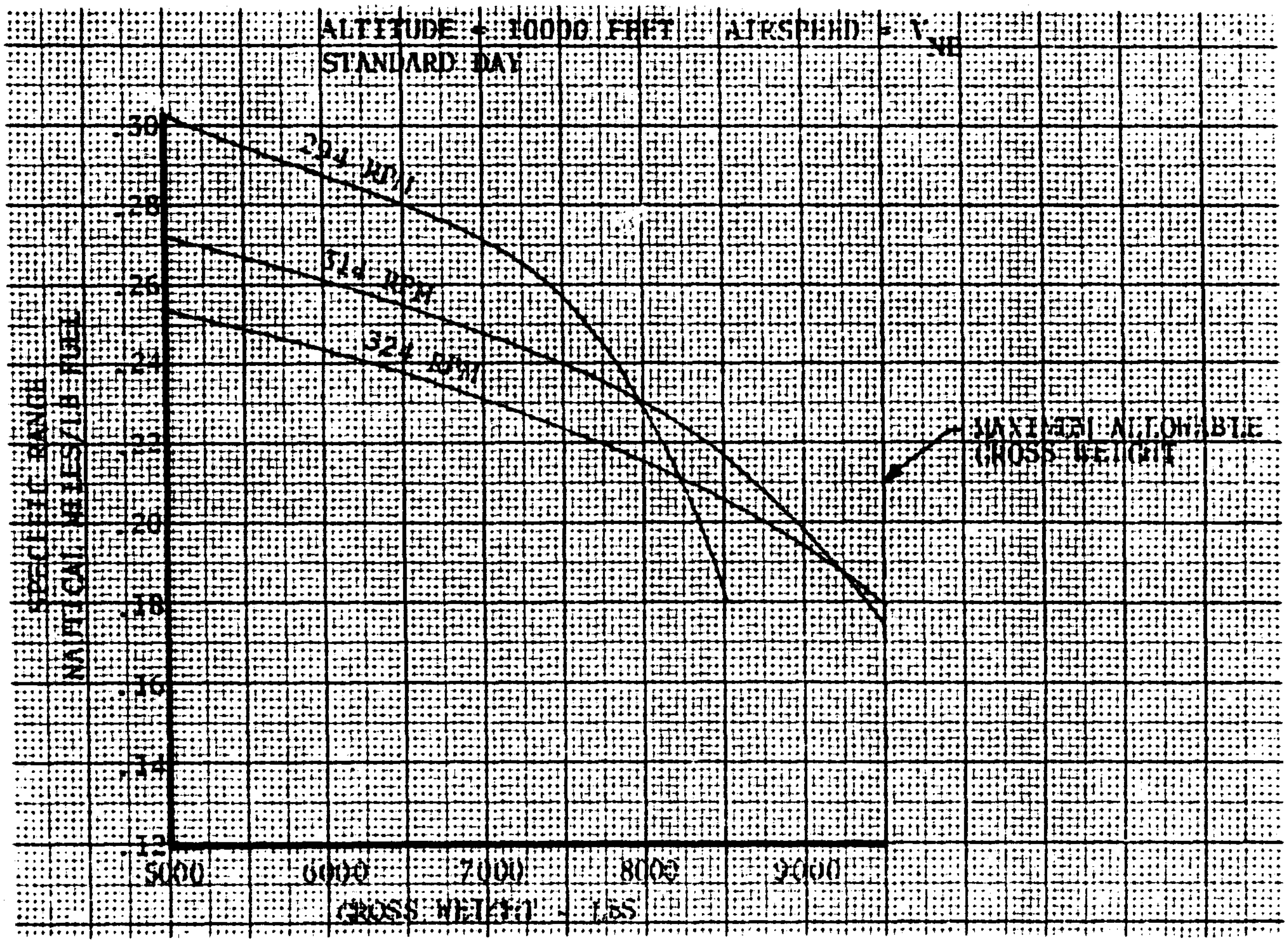

Figure T. Effect of Rotor Speed and Gross Weight on Specific Range.

Optimum Altitude

41. Figure 38, appendix II, shows range performance at optimum cruise altitude. Optimum cruise altitude is that altitude which produces maximum specific range for a given weight. Figure U shows the potential increase in range (in percent) achieved by cruising at optimum altitude as compared to SL: At light gross weights, range can be increased as much as 50 percent by climbing to altitudes as high as 25,000 feet where oxygen is required. A 20- to 30-percent increase can be obtained by limiting the altitude to 10,000 feet where oxygen is not required. Optimum altitude cruise significantly increases the range for ferry flights or any other long range missions.

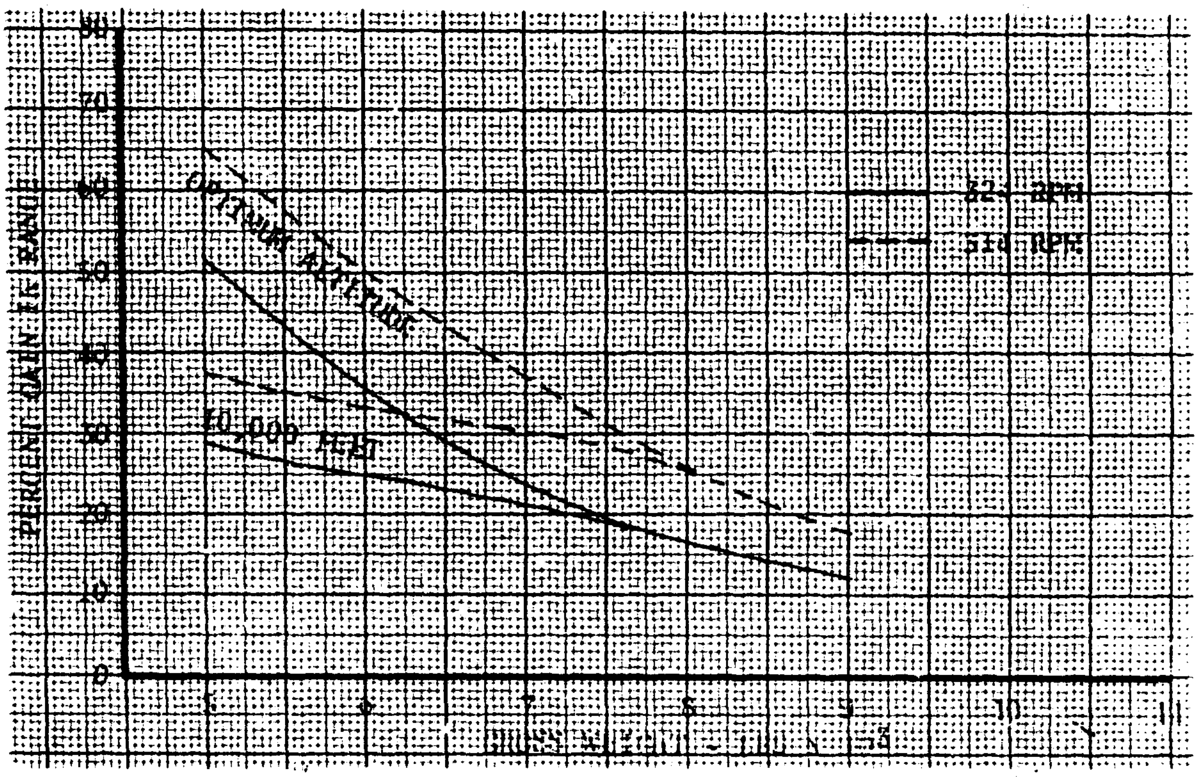

Figure U. Percent Gain in Range at Optimum Altitude Compared to Sea Level.

Endurance:

42. The maximum endurance of the YUH-1H is slightly less than that of the UH-1D. This is because of the increased specific fuel consumption of the T53-L-13 engine at endurance power (fig. 123, app II). Optimum endurance is presented in figures 48 through 56. As with range performance, cold temperatures adversely affect endurance due to compressibility effects. Airspeed has very little effect on endurance over a small range near minimum power speed (maximum endurance speed). Performance presentation is the operator's manual could

be simplified by presenting endurance data only at 45 KIAS. Endurance information is presented at 50 KCAS in figures 57 through 65. These data are conservative but could be used adequately for flight planning purposes.

Airspeed Calibration

43. Airspeed calibration tests were conducted to determine the position errors of both the standard aircraft system and the boom-mounted test system. The tests were conducted at the following conditions: gross weight, 7050 pounds; density altitude, 1130 feet; rotor speed, 324 rpm; cg location, mid; airspeed, 23 to 121 KCAS, level flight; and configuration, clean. The results are presented in figures 83 and 84, appendix II. The boom system position error varied linearly from +4 KCAS at 20 KCAS to +6 KCAS at 115 KCAS. The standard system position error varied from +4 to +10 KCAS nonlinearly. This does not meet the requirements of MIL-I-6115A (ref 13, app I) which stipulates that position error of airspeed indicator systems with flush-mounted static systems cannot exceed ±4 KCAS. The standard system of the test aircraft had flush-mounted static ports and a nose-mounted pitot probe. Later model UH-1D/H helicopters have roof-mounted pitot static probes. The position error of the roof-mounted system as shown in the operator's manual (ref 12) varies from -10 to +6 KCAS. This position error exceeds the limits of MIL-I-5072A (ref 14) which also requires that this error not exceed ±4 KCAS. The position errors of both systems are excessive even when compared to those of other helicopters. The position error of the nose-mounted pitot airspeed system presented in the operator's manual (ref 12) is a constant +4.5 KCAS. Apparently, it was obtained from the test boom installation error of the data for the UH-1D test aircraft (ref 5). This does not agree with the results of this test and could cause a hazardous condition, such as the pilot inadvertantly exceeding V_{NE}. The airspeed installation correction table in all UH-1D/H handbooks for nose-mounted pitot systems should be changed to that of the standard aircraft system presented in figure 83, appendix II. For clarity and to reduce the possibility of confusion, all airspeeds in the operator's manual (ref 12, app I) and elsewhere (V_{NE} placards, etc.) should be presented in terms of indicated airspeed (corrected for instrument error) which is the only parameter directly available to the pilot. For example, the 120-knot red line located on the airspeed indicator is erroneous. The maximum structural airspeed is 120 KCAS which is equivalent to 112 KIAS. This could result in the airspeed limit being exceeded by 8 knots which would reduce the safe life of many airframe components. The red-line limit on the airspeed indicator should be changed on all UH-1D/H aircraft to the correct indicated airspeed.

STABILITY AND CONTROL

General

44. Based on qualitative evaluations, the stability and control characteristics of the test helicopter were acceptable except for operations in the low-speed region where control limitations were encountered. No quantitative stability and control tests have been conducted on the UH-1D/H airframes with the 48-foot rotor by the military other than the low-speed tests of this program. In addition, all qualitative evaluations have been conducted on prototype airframes. Some potentially hazardous conditions may not have been disclosed during qualitative tests. Prototype aircraft may exhibit different stability and control characteristics than production aircraft. Complete quantitative stability and control tests should be performed on a current production model UH-1H.

Dynamic Stability

45. The dynamic stability characteristics were qualitatively evaluated by observing the reaction of the helicopter following a disturbance about the axis being evaluated. During climbs at a mid cg and at all gross weights tested, the test helicopter exhibited longitudinal dynamic instability. The dynamic stability in roll and yaw was positive at altitudes below 10,000 feet. As service ceiling was approached, a continual deterioration was observed about all three axes. The motion about the pitch axis became more divergent. These characteristics caused difficulty in stabilizing at a desired climb condition. In level flight at all gross weights, a mid cg and an altitude of 5000 feet, the motion about all three axes varied from light to well damped. As the altitude and gross weight increased, the motion about the yaw axis became unstable (divergent) below 35 KIAS, and the motion about the pitch and roll axes became unstable at the published maximum airspeed. The dynamic stability characteristics were poor but acceptable.

Static Lateral-Directional Stability

46. The static lateral-directional stability was qualitatively evaluated by observing the control positions and helicopter attitude for various stable sideslip conditions. The static directional stability appeared to be positive for all level-flight conditions tested. Left pedal input was required to sustain right sideslip and vice versa. The static lateral stability (effective dihedral) as evidenced by the lateral cyclic stick position versus sideslip angle was slightly positive at a 9500-pound grwt below 10,000 feet. As gross weight decreased, the effective dihedral decreased becoming slightly negative

at 7500 pounds. Above 10,000 feet, the dihedral effect was neutral at a 9500-pound grwt and negative at 7500 pounds. The helicopter did not meet the positive effective dihedral requirements of MIL-H-8501A for all flight conditions. This characteristic is undesirable but acceptable.

Low-Speed Flight Regime

47. Sideward and rearward flight, landing approaches in wind and hover in wind tests were conducted to determine the capability of the test helicopter to operate in the low-airspeed flight regime (hover, taxi, takeoff and landing). The most significant problem encountered with the UH-1H was the lack of adequate directional control (left pedal) at high power in the low-speed flight regime. This condition also existed with the UH-1D and is aggravated by the increased power available and heavier gross weight operating capability of the UH-1H. It was also determined that an insufficient aft longitudinal control margin exists in the low-speed flight regime. The low-speed flight regime tests were conducted at the conditions shown in table 2. Ballast was added periodically during each flight to maintain the desired center of gravity and gross weight within ±25 pounds as fuel was consumed. This program did not provide for quantitative stability and control instrumentation. An oscillograph for continuous recording of aircraft rates, attitudes, tail rotor power and thrust, which would be required to adequately test for and analyze low-speed stability and control data, was not available. The only stability and control data available were control positions which were recorded at periodic intervals on a photopanel.

Sideward and Rearward Flight:

48. Sideward and rearward flight characteristics were evaluated at the conditions listed in table 2. All ground speeds were corrected to the zero wind condition. Control positions and power required are shown as a function of speed in both sideward and rearward flight in figures 87 through 102, appendix II. Control positions are presented in inches of travel (center of the grip) from full left, full forward or full down. Variation of average pedal position from hovering trim (static) and movement required (dynamic) for each condition are summarized in figure 85 for sideward flight. Figure 86 shows similar information for longitudinal cyclic position in rearward flight. The data were obtained in the following manner: The aircraft was first stabilized on heading and desired speed by reference to a ground pace vehicle. Data were then recorded for approximately 10 seconds while the pilot attempted to maintain speed and heading. Maximum, minimum and stable control positions during this period were read and presented. The stable control position shows the approximate static trim position for the condition. The extreme control positions give an indication of pilot workload and apparent

stability at the condition. At the time of these tests, skid height was not known to be a significant parameter. Therefore, it was not precisely maintained or recorded. Since these tests, skid height was determined to be critical (ref 18). It is probable that some skid height other than that used for these tests is more critical than these results show. Additionally, tests were conducted at relative wind azimuths of only 90 degrees to the aircraft principal axes. It is probable that some wind azimuths other than 90 degrees are more critical as indicated by reference 18 and the hovering and approaches in wind tests conducted during this program. The flight characteristics (pertaining to each axis) encountered during sideward and rearward flight are discussed in paragraphs 49 through 51.

Table 2. Low-Speed Flight Regime Test Conditions.

Rotor speed: 324 rpm Skid height: 10 to 20 feet (except approaches)		Surface winds: zero to 3 KCAS (sideward and rearward flight only)	
Gross Weight (lb)	Longitudinal Center of Gravity (in.)	Lateral Center of Gravity (in.)	Density Altitude (ft)
8,600	130.2 (fwd)	0.0	4,080
9,500	137.1 (mid)	+2.5 (right)	4,250
9,500	134.0 (fwd)	-2.0 (left)	3,750
8,300	136.2 (mid)	0.0	4,840
7,000	130.0 (fwd)	0.0	5,400
8,600	130.0 (fwd)	0.0	12,760
8,600	144.0 (aft)	0.0	4,780
9,500	143.9 (aft)	0.0	5,700

Lateral Axis

49. No control problems were encountered in the lateral axis. Graphical data on lateral cyclic stick position are not presented in this report. Lateral cyclic stick position is presented for four sideward and rearward flights in the interim report (ref 15, app I). Lateral cg location was varied from 2.5 inches right to

2 inches left. This would be the approximate lateral cg with troops loaded on only one side of the aircraft and would be the most extreme lateral cg to be expected with internal loading. With the external rescue hoist, the lateral cg limit of ±7.5 inches could easily be exceeded. A means of obtaining a lateral cg of ±7.5 inches was not available for these tests; therefore, flight characteristics at this loading were not evaluated. When the armored seats are installed, any asymmetrical loads should be located on the right side due to interference of the pilot's right elbow with the armored seats. With a mid lateral cg, lateral cyclic stick was approximately at a mid position in hover. Lateral cyclic position changed approximately three-quarters of an inch for each inch of lateral cg change. In right sideward flight, lateral cyclic position showed a positive gradient (right cyclic with right sideward flight) of an approximate 1-inch displacement for each 20 knots. In left sideward flight, the gradient was essentially neutral with approximately 1 inch of left displacement at 10 knots. These characteristics indicate that lateral control margins may be inadequate at the lateral cg limits. Flight characteristics should be investigated at the lateral cg limits during future tests. In rearward flight, a trend of increasing right-lateral cyclic of approximately 1 inch for each 30 knots was observed. In forward flight, lateral cyclic stick position remained approximately neutral at the speeds tested. Lateral control characteristics were acceptable at all conditions tested.

Directional Axis

50. Directional control problems were encountered at most test conditions. Lack of sufficient directional control was also reported in reference 5, appendix I, for the UH-1D. Directional control limitations were found to be adversely affected as weight, power required or altitude increased or as rotor speed decreased. Quantitative analysis of the data indicates that the UH-1H will meet the directional control requirements of reference 7 up to a referred gross weight (W/σ) of 11,000 pounds at 324 rotor rpm. This is equivalent to a 9500-pound grwt at a 5000-foot H_D. Above these conditions, there is insufficient left directional control. As the altitude increases above 5000 feet or as the rotor speed decreases below 324 rpm, weight must be decreased to retain adequate directional control. The data indicate that a 2-inch right lateral cg decreases the speed in right sideward flight at which the left pedal reaches the stop by as much as 20 knots compared to a mid or left lateral cg (fig. 91). A left lateral cg decreases the right pedal margin in left sideward flight (fig. 89). There is insufficient data with lateral cg offset for any general conclusions. Qualitatively, there was a strong longitudinal cg effect on directional control. At the test density altitudes of 4000 to 5000 feet, there appeared to be adequate directional control at 9500 pounds with an aft cg. As the cg moved to the forward limit, the gross weight had to be

decreased to approximately 8300 pounds to provide adequate control margin. Quantitative analysis of the data does not substantiate this effect. A possible explanation of this apparent discrepancy is the combined effects of imminent loss of both directional control and longitudinal control. This loss of control increased the pilot's workload and divided the pilot's attention. That is, at an aft cg with a large longitudinal control margin, the pilot could devote nearly full attention to controlling the aircraft directionally. However, with the cg at or near the forward limit, the pilot's attention must be divided between controlling the aircraft both in direction and pitch, therefore, making it desirable to have greater pedal margin. It should also be noted that the tests were conducted over unconfined, level terrain (usually a runway) where full attention could be devoted to controlling the aircraft. The consequences of marginal or inadequate directional control were less severe than they would be operationally, where the area may be limited, the terrain may be unsuitable for landing to regain control, or the pilot's attention may be required for tasks other than just controlling the aircraft. Inadequate directional control generally caused the aircraft to turn toward the relative wind. Several times, the aircraft continued a right turn through the relative wind azimuth at a moderate rate of turn even with full left pedal continuously applied. The only method of recovery was to reduce collective (power) and allow the aircraft to settle to a low skid height or to contact the ground. Lack of sufficient directional control throughout the weight, altitude and cg envelopes of the UH-1H adversely affects safety of flight and limits mission capability. This information should be included in chapters 7 and 8 of the operator's manual. The right pedal stop was contacted at several conditions at speeds above 30 knots in left sideward flight. An additional problem encountered in left sideward flight was a discontinuity in pedal requirement of approximately 3 inches between 10 and 20 knots to the left. This pedal discontinuity occurs at approximately zero tail rotor blade angle. There was difficulty in maintaining heading and an apparent instability in this airspeed range. Associated with this discontinuity was high directional sensitivity, approximately twice as sensitive as it would be without the discontinuity. This high sensitivity may have contributed to the apparent yaw instability encountered in forward flight (para 45). An estimate of pedal effectiveness versus displacement based on pedal requirements during hover, sideward and rearward flight, and operations in wind is presented in figure V. Desired effectiveness is also presented.

Longitudinal Axis

51. Insufficient aft longitudinal control power was experienced during the testing. Increased gross weight, power required, and forward longitudinal cg adversely affected longitudinal control.

Insufficient longitudinal control was also reported in reference 5, appendix I, for the UH-1D. Within 3 inches of the present forward cg limit, sufficient aft longitudinal control margin was not available for control of the YUH-1H in rearward flight. As the cyclic stick reached the aft stop, there was an uncontrollable nose-down pitching motion. Where directional control and power available permit, recovery can be accomplished by turning into the relative wind and/or adding power to gain forward speed. This method is not recommended because adequate directional control may not be available; and the additional power, if available, would adversely affect directional control. Recovery can also be accomplished by reducing power and settling to the ground. On several occasions, simultaneous lack of directional control and longitudinal control resulted in an uncontrolled right turn and a nose down pitch rate which were arrested only after reducing collective and settling to the ground. On two occasions, the loss of control was severe enough that entry into autorotation was contemplated to regain control. Lack of sufficient aft longitudinal control adversely affects flight safety.

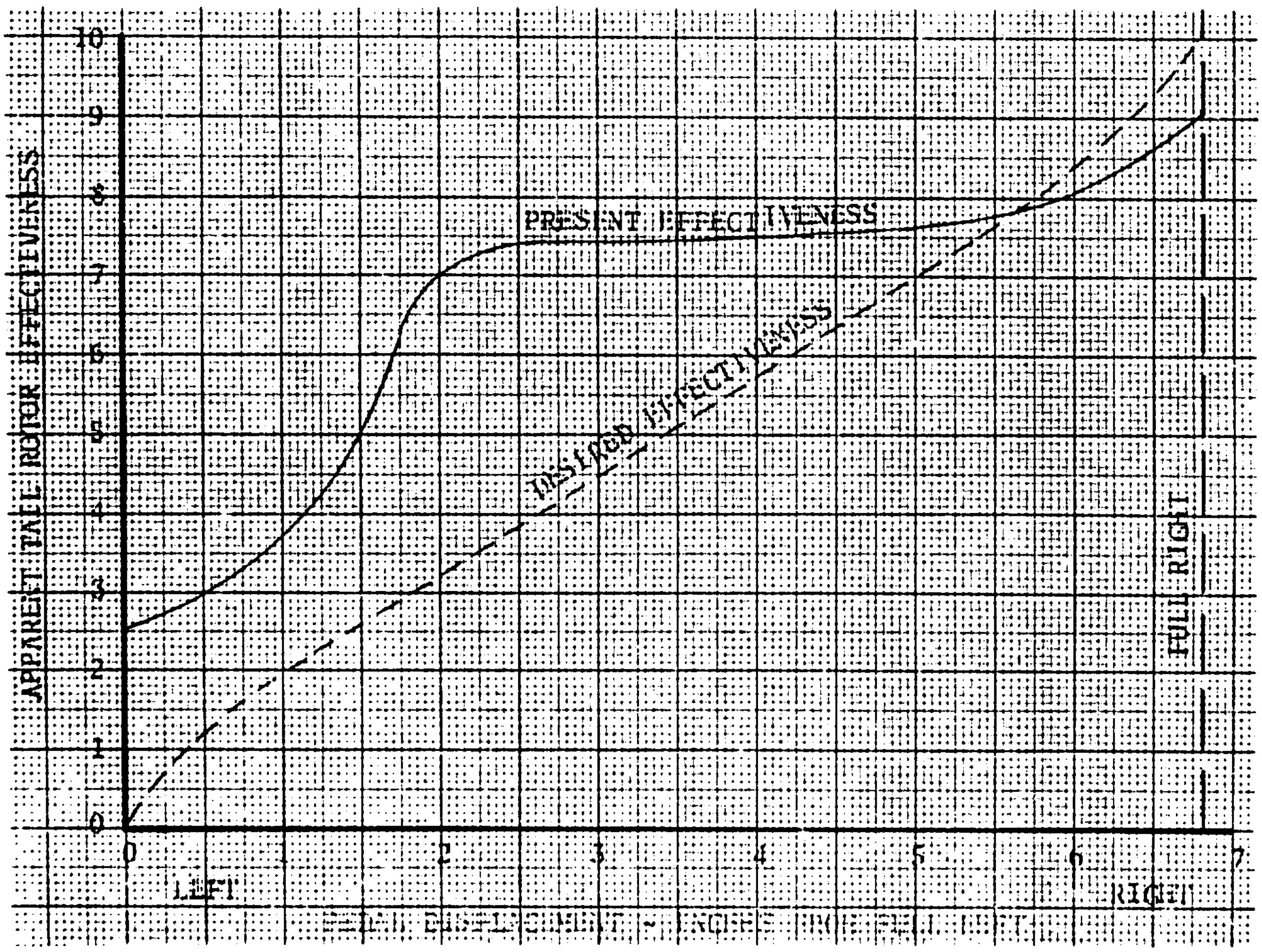

Figure V. Estimated Tail Rotor Effectiveness.

Hovering in Winds:

52. Hover tests were conducted in winds to determine the adequacy of sideward and rearward flight tests to simulate operations in winds. The tests were conducted at the conditions as shown in table 2 except that wind speeds were from 5 to 18 knots. Figures 103 through 105, appendix II, show examples of control positions while hovering in winds at various azimuths. Actual hovering in winds was determined to be more critical than sideward and rearward flight. While hovering in winds at conditions where less than 10-percent average pedal control remained in sideward and rearward flight, full control would be reached, and the turn could not be completed. At a forward cg while attempting to hover downwind, the aft control stop was reached, and a pitch attitude or position over the ground could not be maintained. At conditions where high power (above a 40-psi torque) was required to hover, insufficient left pedal control was available. Any yawing moment generated by a gust could not be controlled. As rotor speed was decreased below 324 rpm, directional control became more critical. These characteristics are detrimental to flight safety.

Approaches in Winds:

53. Approaches were made at various wind azimuths to determine the adequacy of the sideward and rearward flight tests to simulate operations in surface winds. Examples of control positions, power parameters and flight path data are shown in figures 106 and 107, appendix II. Other examples are shown in the interim report (ref 15, app I). Control problems were less severe while making approaches in winds than while hovering in winds or during sideward and rearward flight. Winds from the right rear quadrant were determined to be the most critical for both longitudinal and directional control. Steady left pedal requirements were greater for either right crosswinds or right quartering tail winds than for the other wind azimuths. However, downwind approaches required more and larger collective changes to maintain glide slope than other azimuths; therefore, more and larger transient pedal inputs were required. It was also found that with a tail-wind component, nose-down pitch rates must be kept small when leveling after the landing flare to avoid contact with the aft stop and loss of longitudinal control. Several times it was necessary to apply power before the aircraft had stopped and make a go-around due to uncontrollable nose-down pitch rates. At an aft cg (aft of station 140), regardless of winds, it was necessary to terminate the approach at a minimum skid height of 5 feet to keep the tail skid from contacting the ground during the landing flare due to tail-low attitude. During approaches in gusty wind conditions, rotor speed varied more than ±5 rpm because of the poor static droop characteristics (para 71). This variation either caused transient rotor overspeeds, or it further compromised

directional control if rpm was "beeped" down. These characteristics are detrimental to flight safety. At the 8- to 10-degree approach angle used, a 2- to 4-psi transient torque increase over that required to hover at 2 feet was needed to terminate the approach. If steeper approaches were made or if large control or flight path excursions occurred, this transient torque increase became much larger.

Analysis of Low-Speed Flight Regime:

54. In the absence of adequate instrumentation or quantitative stability and control data on the UH-1D/H aircraft with the 48-foot rotor and the lack of knowledge of the significant effects of skid height and wind azimuth during these tests, a precise analysis is not possible. However, there is no question that the UH-1H helicopter does not meet the longitudinal and directional control requirements of paragraphs 3.2.1, 3.3.2, 3.3.5, and 3.3.6 of reference 7, appendix I, even as amended by deviation 7 of reference 17 which reduces the airspeed/wind-speed requirements from 35 to 30 knots. This is evident from the fact that full control (on the stop) was reached within the rotor speed, loading (weight, longitudinal and lateral cg), altitude and power limits (para 3.1.2, ref 7) at speeds less than 30 knots. Therefore, no control margin existed, regardless of interpretation of specification requirements. It is also obvious that during normal operation at some conditions within the operational envelopes (altitude and loading) of the UH-1H, there are insufficient control margins, and control of the aircraft will be lost. The following analysis was made in an effort to determine at what conditions control would become marginal or inadequate. Reference 18 has shown directional control requirements to be a function of referred gross weight (gross weight/density ratio - W/σ) at a given rotor speed. Hovering trim pedal positions varied linearly from a pedal position of 0.8 inch from full left at a W/σ of 12,700 pounds to 2.6 inches at a W/σ of 8200 pounds. The average pedal trim change from zero to 35 knots right sideward flight was 0.8 inch. In the absence of controllability data, a 10-percent pedal displacement requirement for maneuvering, to counteract gusts and basic aircraft instability, was assumed. This gives a 1.48-inch pedal displacement requirement from hovering trim to permit sideward flight to 35 knots and to allow for maneuvering. This pedal displacement requirement corresponds to a W/σ of 11,000 pounds. Although no qualitative data were recorded at rotor speeds lower than 324 rpm,, reference 18 indicates that directional control is a function of main rotor C_T. Using this information, an estiamte of the W/σ for adequate directional control at other rotor speeds gives a W/σ of 10,330 pounds for 314 rpm and a W/σ of 9060 pounds for 294 rpm. Due to the poor static droop characteristics (para 71), it is impossible to maintain the rotor speed at 324 rpm for all low-speed operations; therefore, the decreased directional control margins with lower rotor

speeds will be encountered in normal service use. Additionally, at some conditions, hover performance is improved at lower rotor speeds making it desirable to operate at rotor speeds lower than 324 rpm. The recommended envelope of weight and density altitude based on this analysis is shown in figure W. For longitudinal control, reference 7 requires that 10 percent of the maximum attainable pitching moment in a hover be available to control the effects of longitudinal disturbances. In the absence of controllability data, this was interpreted to mean 10 percent of the total control travel or 1.22 inches. At the forward cg limit, this margin is not available as shown in figure 88 at a weight of 8600 pounds and figure 92 at a weight of 9500 pounds. Figure 90 shows that at a weight of 9500 pounds and at a cg of 137.1, the UH-1H just meets the 10-percent margin. At a gross weight of 8300 pounds, the 10-percent margin is just met at a longitudinal cg of 133.0. This was determined by subtracting the hovering trim shift of 0.9 inch, as the cg moves from 136.2 to 133.1, from the margin of 2.2 inches at 35 knots right sideward flight at a cg of 136.2 as shown in figure 92. In both cases, the required margin is available at a longitudinal cg 3 inches aft of the present forward limit. Therefore, the UH-1H should be limited to a longitudinal cg 3 inches aft of the present forward limit to insure adequate control margin for operations in surface winds (low-speed flight regime) as shown in figure 130, appendix II. Since these envelopes restrict the operational capability of the UH-1H, it is recommended that adequate directional and longitudinal control for safe operation within the current envelope be provided.

55. Control requirements during tests in actual surface winds (paras 52 and 53) were more severe than during sideward and rearward flight at the same conditions. At conditions where the average control position in sideward and and rearward flight was within the 10-percent control margin, control was lost while hovering or making approaches in actual winds. At conditions where the maximum excursion of the control position in sideward and rearward flight did not penetrate the 10-percent margin, control was adequate. This indicates that the control margin requirement should be applied to the maximum excursion in the critical direction during sideward and rearward flight rather than the trim position at sideward or rearward flight speeds. The maximum wind speed during the tests in wind was 18 knots, well below the specification requirement of 30 knots. It is recommended that further tests be conducted on a production model UH-1H to determine the critical wind azimuth, the effect of various skid heights, and the control margins required for safe service operations.

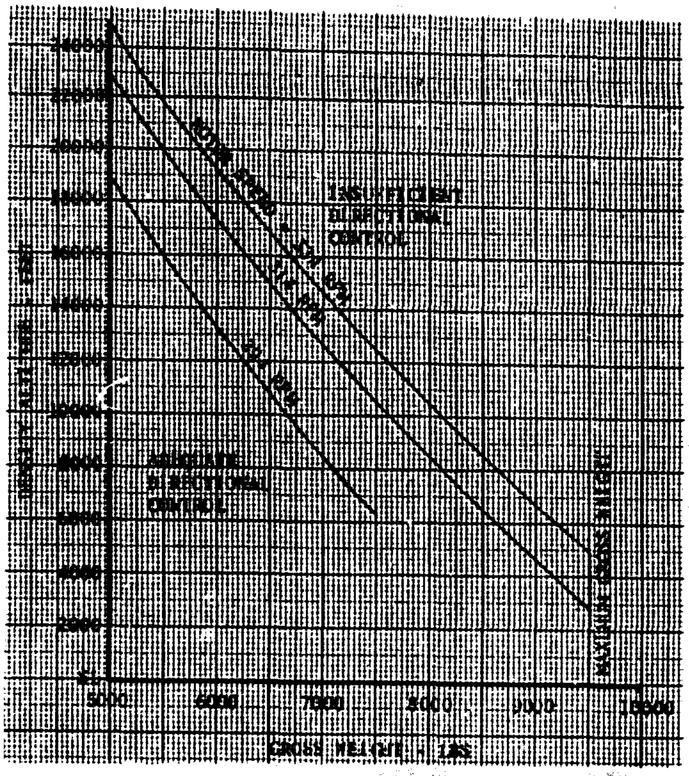

Figure W. Recommended Weight - Altitude Limits for Adequate Directional Control.

ENGINE PERFORMANCE

Power Limits

56. The T53-L-13 engine is rated at 1400 shp for SL, standard-day uninstalled conditions. It will produce 1340 shp at SL, standard-day conditions as presently installed in the UH-1H. However, the UH-1 transmission has a maximum torque rating of 905 ft-lbs, which limits the maximum allowable power to 1137 shp at 6600 engine output shaft rpm (324 rotor rpm) or to 1102 shp at 6400 engine output shaft rpm (314 rotor rpm). Ratings and limits stated in the operator's manual (ref 12, app I) disagree with these limits. When the nominal specification torque constant is applied to this 905 ft-lb torque limit, the resultant indicated torque pressure limit is 49 psi, not 50 psi as stated in the operator's manual. In addition, when the allowable accuracy tolerance of the torque system is applied, the indicated limit pressure may vary from 45 to 53 psi. The result of this inaccuracy can be either a loss of a useful hover load up to 600 pounds or a transmission overtorque of 9.5 percent. Although the torque system of each test engine was accurate (maintained individual calibration), the variation between test engine torque systems was near the maximum allowable. Sufficient information

is available on each engine data plate to derive an accurate indicated torque pressure limit. One other factor which degrades the accuracy of the torque system is the poor readability of the torque gage. In view of the critical nature of the torque limit, both the torque system accuracy and the torque gage readability are considered unacceptable. The allowable power-on rotor/engine transient overspeed limit (331 rotor rpm/6750 engine rpm) as stated in the organization maintenance manual (ref 16) should also be stated in the operator's manual (ref 12). This transient overspeed is required because of the poor static droop characteristics and the detrimental effect of low rotor speed on directional control. In addition to the problems encountered with the mechanical ratings and limitations, one problem was encountered with the engine thermodynamic limits. A steady state, fuel-flow limiting device (N_1 topping system) automatically establishes military rated power (MRP) when the twist grip is fully open. Maximum continuous power is established by monitoring the exhaust gas temperature (EGT). However, on the test engines used, maximum continuous power could be exceeded by as much as 30 percent without exceeding the maximum continuous EGT limit of 625°C. This may have contributed to the engine failures experienced (para 73). A better method of determining the maximum continuous thermodynamic power limit is to subtract a 2-percent gas producer speed (N_1) from the N_1 trim value.

Installation Losses

57. Engine performance ratings and guarantees are established for an uninstalled engine at optimum, static, SL, standard-day conditions. When the engine is installed in an airframe, the following items cause a decrease in engine output shp available and an increase in fuel consumption: engine air inlet ducts, engine air inlet filters, exhaust ducts, exhaust-type heaters, infrared suppression exhaust devices, gas producer section power extractions for driving accessories, compressor bleed air extraction, inlet anti-icing equipment, and operation at nonoptimum turbine and compressor speeds, etc. Methods of correcting engine performance for the losses caused by these items are included in the engine model specification. For the T53-L-13 engine as installed in the UH-1H, the loss in MRP available was 4.3 percent from 1400 output shp to 1340 shp at SL, standard-day conditions. Paragraphs 58 through 64 discuss each installation loss and its effect on engine performance.

Engine Air Inlet Losses:

58. Five different inlet configurations were tested to determine their effect on performance. All configurations are presently used on various UH-1D/H aircraft with the exception of the

particle-separator/screen combination which was intended to simulate removal of the fiber filler from the barrier filter. Each inlet configuration is described in appendix IV. The results of the tests are shown in figures 114 and 115, appendix II.

59. Engine air inlet temperature rise varied from -2° to +3.5°C with the average rise being approximately +1°C. This rise was used for all engine performance determinations. There was no discernible trend with inlet configuration, airspeed, altitude, power or dirtiness of the filters. There was no apparent trend of additional increasing temperature in a hover except for the direct downwind hovering condition, and this trend was transient in nature. The variation of inlet temperature rise had the same effect on both power and fuel flow as an ambient temperature change.

60. Engine inlet air pressure ratios (P_{T_2}/P_{amb}) varied from a maximum of 0.995 with the bellmouth louvers configuration in a hover, to a minimum of 0.97 with the particle-separator/barrier-filter configuration at maximum forward speed. The trend of all configurations was a decreasing pressure ratio with forward speed. The external louvers minimized this decrease. The particle separator with either the louvers or the barrier filter produced the lowest ratio in a hover and at low airspeed. Since the most critical condition of power available is hovering, the worst configuration in a hover was used for all engine performance determinations. This configuration was the particle-sepaator/barrier-filter combination which had a pressure ratio of 0.987 in a hover. Normal power available data in forward flight for this configuration (fig. 119, app II) shows a power decrease of 31 horsepower at maximum forward speed which is the worst condition. Rotor speed or altitude change had no effect on inlet pressure ratio. Pressure ratio decreased slightly in climbing flight, possibly due to lag in the static system or because of increased air flow to the engine at higher power settings. The barrier filter was allowed to become visibly dirty, and the internal filter of the particle separator became completely blocked. No noticeable pressure drop occurred with the filters dirty. If the filters become completely blocked with grass or ice, a large pressure drop could be expected. Until the barrier filter clogged light activates, pressure drop will be less than 1.5 percent from the best inlet condition, and power loss will be less than 25 shp. With all configurations, pressure drop due to the inlet configuration was less than the equivalent of a 600-foot increase in altitude. The difference from the best configuration to the worst configuration was less than the equivalent 300-foot difference. This pressure loss had no significant effect on fuel flow. The effect of inlet pressure ratio change on power available is shown in figure X at one condition.

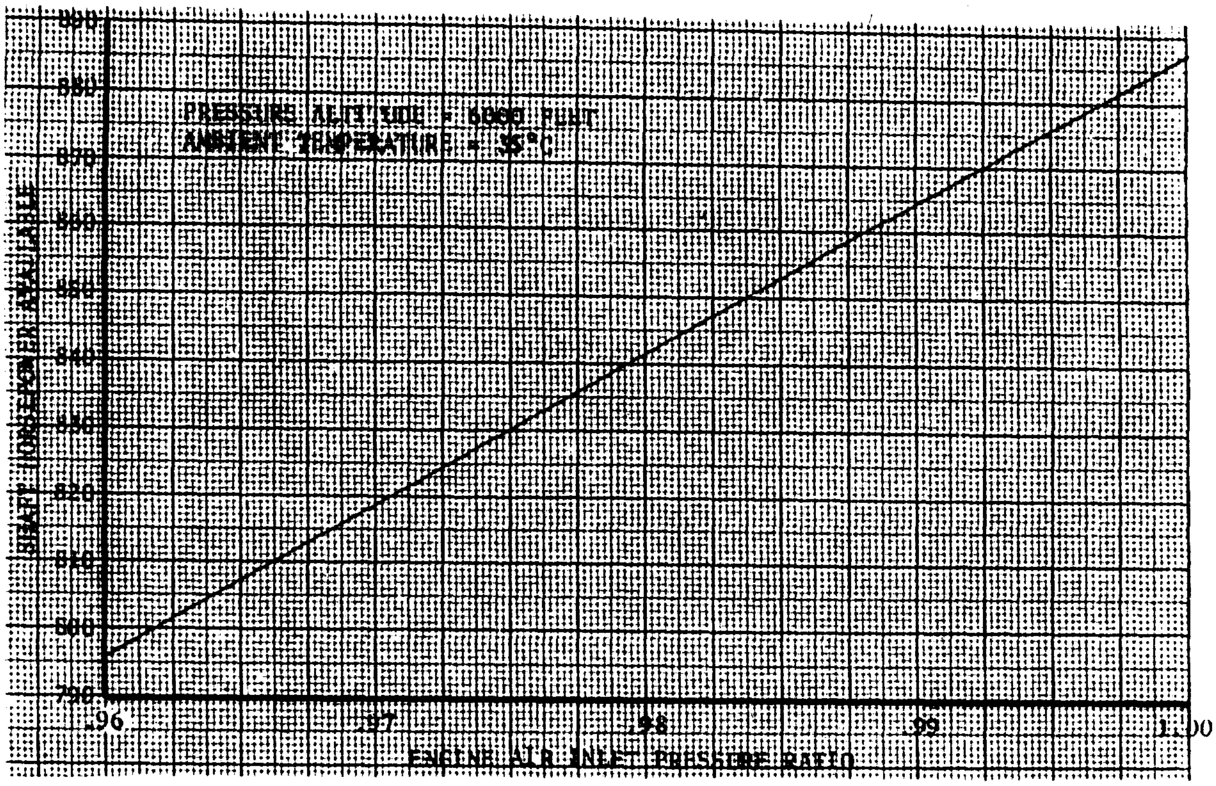

Figure X. Effect of Inlet Pressure Ratio on Power Available.

Compressor Air Bleed:

61. The quantity of bleed air used was not measured during these tests. The results and analysis of the airframe contractor's flight test show that a constant 1.15 percent of the total air flow was used to run the fuel boost pump and oil cooler fan. This value was used for engine performance determinations when the bleed air heat or engine anti-icing system was not used. When the bleed air heat was used, maximum available bleed air of 4 percent of the total air flow was used for performance determinations. With the anti-icing ON, 0.5 percent of the total air flow from this 4-percent maximum air bleed is diverted through the engine inlet anti-icing system leaving only 3.5 percent for other functions. The worst condition of bleed air extraction is with the anti-icing ON since the 0.5-percent flow for anti-icing is diverted back through the compressor effectively causing an inlet temperature rise. The effect of bleed air on power and fuel flow is shown in figures Y and Z.

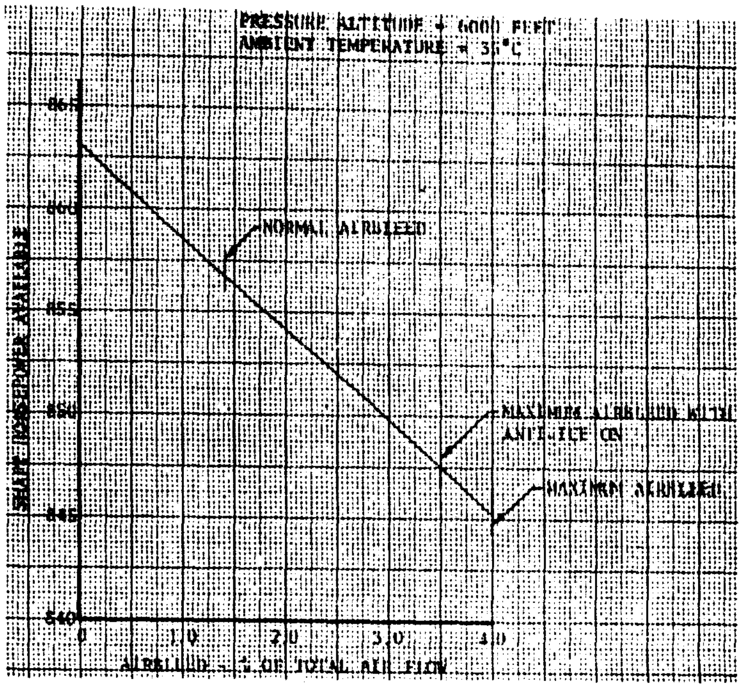

Figure Y. Effect of Air Bleed on Power Available.

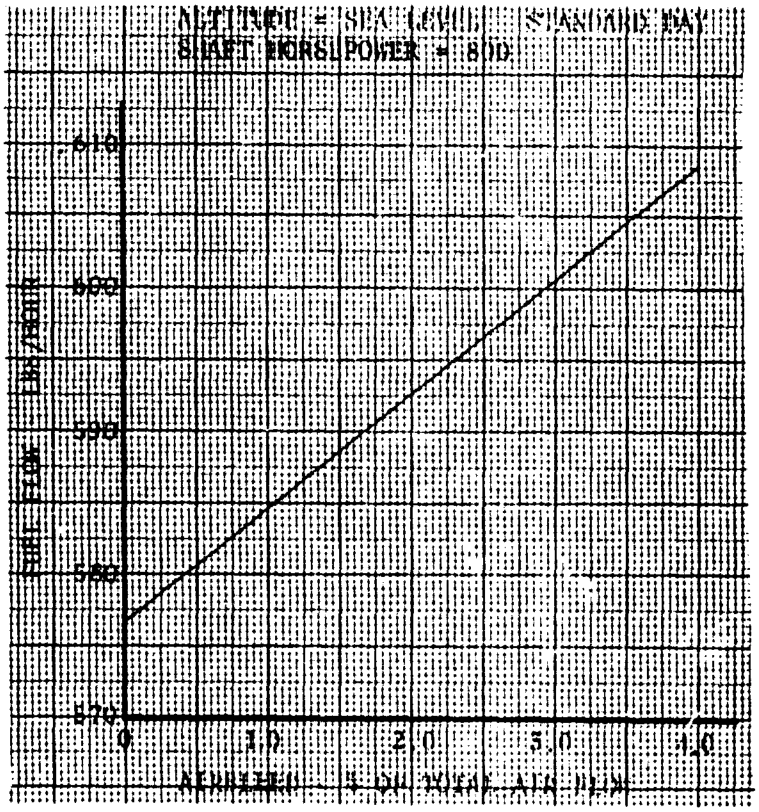

Figure Z. Effect of Air Bleed on Fuel Flow.

Power Turbine Speed:

62. Power turbine speed has a significant effect on power available and fuel flow. Generally, optimum turbine speed fell below the normal, minimum allowable power-on rotor speed of 314 rpm (6400 output shaft rpm/20,547 power turbine rpm). However, most performance was determined at a maximum steady state rotor speed of 324 rpm (6603 output shaft rpm/21,200 power turbine rpm) for considerations of rotor performance, aircraft control or vibration. In addition, at the 905 ft-lb torque limit, 35 additional horsepower were available at the maximum rotor speed. Effects of turbine speed on power available and fuel flow are shown in figures AA and AB for one condition.

Gas Producer Power Extraction:

63. Power extracted from the gas producer section results in less power available from the power turbine output shaft. The power extracted usually drives a starter/generator. Since the main generator is driven from the power turbine train on the UH-1H, the starter/generator is considered to be a standby generator, and no electrical load was assumed. An added starter/cooler fan was considered. Per the airframe manufacturer's analysis, a constant 0.4 horsepower extraction was used for the starter/cooler fan. The power required for the main generator is included in the power required data. This is approximately 8 horsepower.

Exhaust Losses:

64. The tailpipe of the UH-1H is the same, both in area and in configuration, as the standard tailpipe of the T53-L-13. Exhaust losses were zero relative to a specification engine. Muff heaters or infrared suppression tailpipes would cause some exhaust losses.

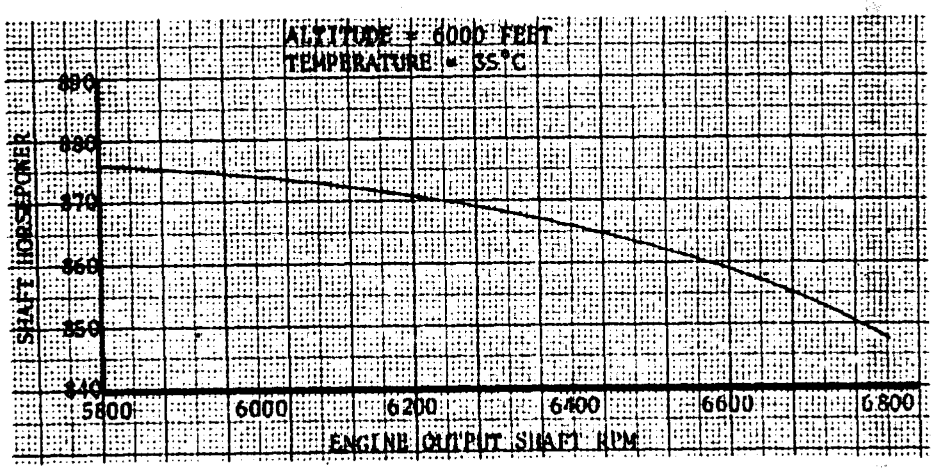

Figure AA. Effect of RPM on Power Available.

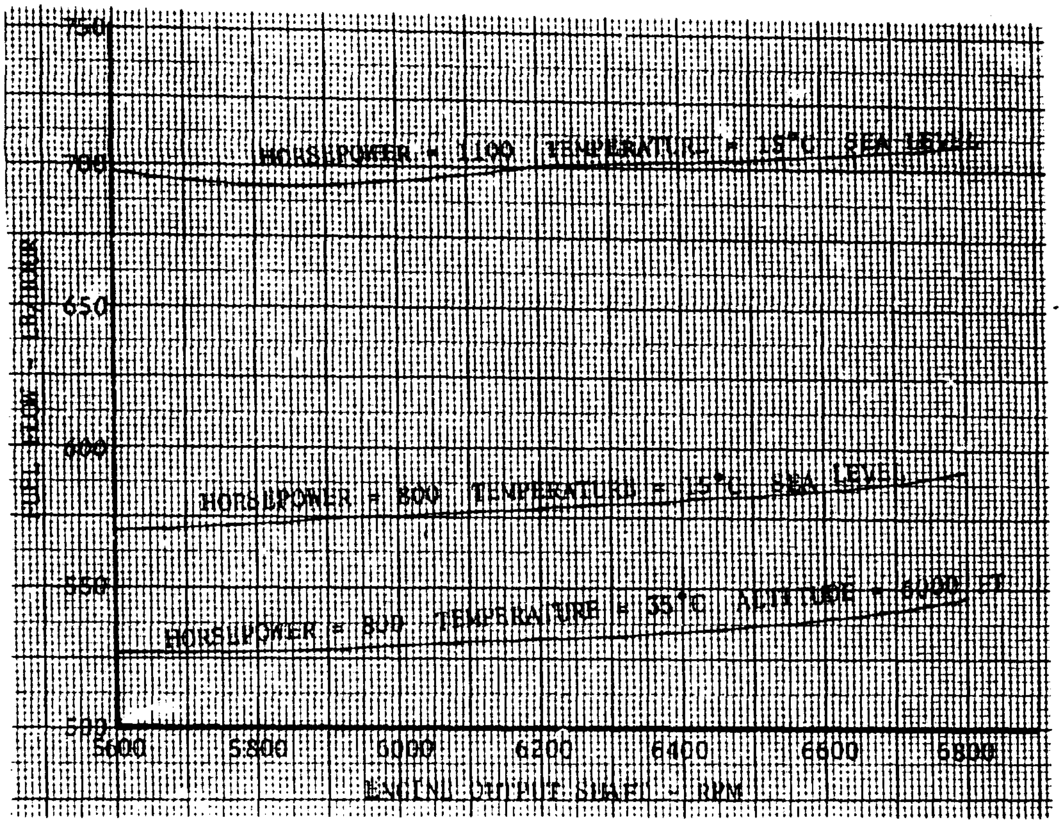

Figure AB. Effect of Engine Speed on Fuel Flow.

Power Available

65. Power available from the T53-L-13 engine, as installed in the UH-1H, was derived from the engine model specification and corrected for installation losses. The following installation conditions were applied: inlet temperature rise, +1°C; inlet pressure ratio, 0.987; N_1 power extracted, 0.4 horsepower; compressor bleed air extracted, 1.15-percent total air flow; exhaust losses, zero; and anti-icing, OFF. When power was computed for cold temperature operation, bleed air was increased to 3.5 percent and the anti-icing was assumed to be ON. These installation conditions are discussed in paragraphs 57 and 64. Power was generally computed for a power-turbine speed equivalent to 324 rotor rpm. Regardless of power setting (military or maximum continuous) or rotor speed, power was limited to that equal to a 905 ft-lb engine output shaft torque because of airframe drive train limits (1137 shp at 324 rpm, 1102 shp at 314 rpm). Power available as a function of pressure altitude and ambient temperature for various power settings and rotor speeds is shown in figures 116 through 118, appendix II. Maximum continuous power with the anti-icing and bleed air heat ON is shown in figure 120. The effect of airspeed on maximum continuous power is shown in figure 119.

Variation of Engine Performance:

66. All performance calculations (range, climb performance, takeoff performance, etc.) were based on the engine model specification and corrected for the installation losses as listed in paragraph 65. The power available calculations were made at the specific ambient conditions at which the performance was computed. This is conservative since all engines when delivered are trimmed to produce at least specification power. The test engines used were trimmed from 2 to 10 percent above specification power. This variation in original power available, the inaccuracy and ambiguity of the various power limits and the probable change in performance with engine condition (time) all cause substantial variation in performance among aircraft and under some conditions will reduce engine life. A procedure for periodic retrimming of the engines to specification power (as closely as practical) would have two potential advantages which are as follows:

a. All aircraft would deliver approximately the same performance, thus eliminating performance uncertainties between particular aircraft.

b. Performance information presented in the operator's manual and elsewhere would be more valid and reliable.

Comparison of Engine Models:

67. "Critical" altitude or temperature is defined as that altitude or temperature where thermodynamic and mechanical power limits coincide. The various engine models used in the UH-1D/H model airframe, as well as test engines used for this program, are compared in figure AC by presenting the pressure altitude and ambient temperature at which they produce UH-1H mechanical limited power (905 ft-lb at 6600 output shaft rpm). In addition, the takeoff power rating (5-minute limit T_{t_5} = 1795°F) of the Lycoming Model LTC1K-4C engine is included since two of the test engines were trimmed slightly above this power. The UH-1H, therefore, has been essentially evaluated at this power setting. At high temperatures, where directional control or torque limitations are not encountered, this power setting would significantly improve hover performance. For example, at the 6000-foot, 35°C condition, the UH-1H would be able to hover OGE at an approximate 8400-pound grwt at takeoff power as compared to a 7400-pound grwt at MP (maximum approved power setting). If this power setting was approved and directional control was adequate, the performance of the UH-1H would not be limited by power available at any normal condition.

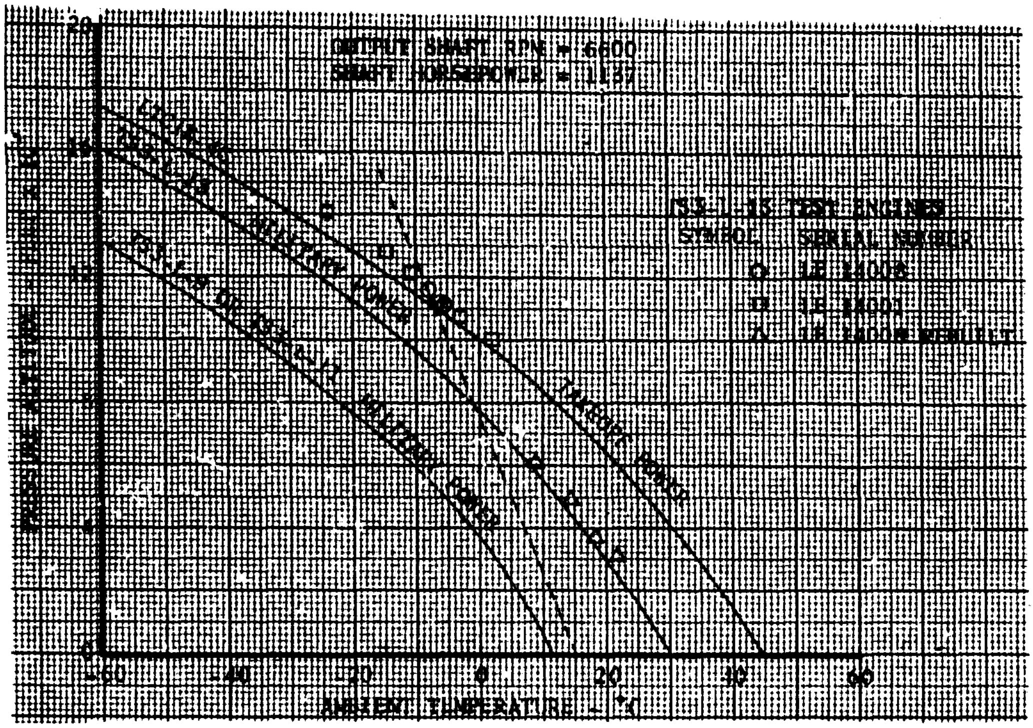

Figure AC. Critical Altitude and Temperature as Installed in YUH-1H.

Fuel Consumption

68. Fuel consumption characteristics of the T53-L-13 engine were derived from the engine model specification and corrected for installation conditions listed in paragraph 65 and discussed in paragraphs 57 through 64. Fuel flow (standard day) is presented in figure 121, appendix II, as a function of power and altitude. Fuel flow with the anti-icing and bleed air heat ON is presented in figure 122. All performance calculations were based on specification fuel flow corrected for installation losses and the ambient condition at which the performance was computed. A direct comparison between test engine fuel flow and specification fuel flow is made on the specific range portion of each level-flight plot (figs. 68 through 82). The points presented are based on test engine fuel flow, and the fairings are based on specification fuel flow on a standard day. The specification fuel flow was 2- to 4-percent conservative as compared to any test engine used. No additional conservatism was applied to specification fuel flow. As operation time is accumulated on an engine, fuel consumption may increase. Fuel consumption on several "high-time" engines should be determined to find the magnitude of fuel consumption change versus running time. The fuel consumption characteristics of the T53-L-13 may be compared to the T53-L-9 by referring to figure 123 which shows a specific fuel consumption

comparison. Below 700 referred shp, the T53-L-13 uses more fuel per shp; above 700 referred shp, the T53-L-13 uses less fuel per shp than the T53-L-9 engine.

ENGINE OPERATING CHARACTERISTICS

Engine Starting

69. Satisfactory engine starts were accomplished at density altitudes up to 14,000 feet. Battery starts could be accomplished within the starter limit of 40 seconds if the battery no-load voltage was 22 volts or greater. With a fully charged battery (24 volts), the starter engaged time varied from 24 to 35 seconds with the initial starter load dropping battery voltage to 16 volts. It was found that the T53-L-13 engine start procedure could be separated into two phases. This procedure differs from that outlined in the operator's manual (ref 12, app I), but the maximum exhaust gas temperature (EGT) during starting is approximately 100°C lower. The two-phase starting procedure outlined below would reduce excessive EGT during engine starts. This procedure should decrease the number of hot starts and did allow satisfactory starts at higher altitudes, higher ambient temperatures and under poor engine or battery conditions.

a. In the first phase, the throttle is positioned at the ground idle/start position detent (49- to 51-percent N_1), and the starter/trigger switch is engaged. A "light-off" of the engine will occur at approximately 12-percent N_1 indication. (On the test engine, the fuel control ground idle/start detent could be felt with difficulty through the throttle linkage. A more positive means of determining the ground idle detent should be provided.) The starter/trigger switch is released at an N_1 indication of 42 percent with the engine accelerating to ground idle. Upon completion of this first phase, the N_1 rpm stabilizes at approximately 51 percent. Elapsed time for this phase varies from 24 to 35 seconds. Maximum observed peak EGT's were 600°C (cold engine) and 680°C (warm engine). EGT's of 675° to 760°C are permitted for a maximum of 5 seconds.

b. In the second phase of the engine start, the throttle is advanced from the ground idle/start position to the flight idle position (70- to 72-percent N_1). Acceleration times and peak EGT's depend on technique and ambient temperatures. Abrupt throttle movements should be avoided at all times.

Engine Dynamic Stability

70. Dynamic stability characteristics of the T53-L-13 engine were qualitatively investigated and appeared to be good throughout the flight envelope tested. Engine response was faster than that of

previous model T53 series engines. When rapid power demands were made, no tendency for compressor stall was experienced with the faster acceleration rates. Power overshoot was small and damping was good. An undesirable engine oscillation (hunting) existed when the engine was topped (N_1 limit). This condition presented itself as a rapid power change at a frequency of approximately 2.5 hertz which produced a one-half-per-rotor-revolution airframe rock. An example of this oscillation is shown in figure 113, appendix II. All parameters were recorded on photopanel instruments and, as such, were not suited for analyzing the instability problem. Gas producer speed or EGT oscillations were not noticeable; however, the instruments may not have been sufficiently sensitive to respond to oscillation. These oscillations appeared to be greater as the altitude decreased (power and ambient temperature increased). At one condition of high forward airspeed, the indicated torque pressure oscillations exceeded ±5 psi. The airframe manufacturer has analyzed this oscillation and stated that it will not cause damage to the drive train or rotor components. A decrease of ½- to 1-percent below maximum N_1 speed eliminated this undesirable characteristic; however, it also reduced power and, therefore, degraded the mission effectiveness of the helicopter.

Engine Static Stability

71. Static stability (rpm droop) of the T53-L-13 engine/airframe combination was unsatisfactory. The change in rotor speed versus engine power output was objectionable for all flight conditions tested. The increased aircraft loading/ambient condition flight envelope aggravated this situation. Constant manipulation of the power turbine speed-select "beep" switch was required to maintain a desired rotor speed during power changes. For example, when making a descent by reducing engine power from a cruise setting of 1007 shp to 107 shp, rotor speed increased 17 rpm (5 percent) when the "beep" switch was not used. When operating at 324 rotor rpm (6600 power turbine speed (N_2)), an engine overspeed resulted. During approaches in gusty wind conditions, the rotor speed varied ±5 rpm. Undue pilot attention was required to maintain a constant rotor speed resulting from the installed droop compensator not being compatible with the engine/airframe combination. The poor static droop characteristics could have an adverse effect on the mission effectiveness of the UH-1H helicopter. The engine static stability defined by rotor rpm droop for various flight conditions is presented in figures 108 through 112, appendix II. Prior to conducting these tests, the droop compensator cam was rigged to the maximum compensation allowed by the organization maintenance manual (ref 16, app I). The droop compensator should provide the best possible compensation for all practical conditions of weight, altitude, power and airspeed with a maximum of two rotor rpm overcompensation for any power change. The present compensator does not maintain N_2 speed at the rpm value selected by the pilot

throughout the flight envelope. A droop compensator cam should be specifically designed for the UH-1H to meet these requirements.

Maximum N_1 Speed Adjustments

72. The procedures outlined in the organization maintenance manual (ref 16, app I) for ground adjustment of the gas producer (N_1) maximum speed are not valid for the T53-L-13 engine. The T53-L-13 engine N_1 speed at minimum collective position is higher than that of previous series T53 engines (82 to 86 percent). Because of this condition and the torque limit, the T53-L-13 engine maximum N_1 speed can only be checked in flight. A method should be provided for conducting ground checks and adjusting the maximum N_1 speed of the T53-L-13 engine. At delivery, two engines used for this program were trimmed approximately 10 percent above specification power by the engine manufacturer.

MISCELLANEOUS

Engine Failures

73. Two T53-L-13 engine changes were required during the test program. Both of these incidents were caused by gas producer (N_1) nozzle failures resulting in loss of power in flight. Total time on engine S/N LE14008 was 16 hours, 50 minutes; on engine S/N LE14001, the engine time was 83 hours, 55 minutes. Damage to engine S/N LE14008 is shown in photos 6 and 7. These engine failures were attributed to incorrect machining of the nozzle assembly. Approximately the first 50 production T53-L-13 engines had a machined nozzle assembly. The engine manufacturer has reported no failures of the later cast nozzle assemblies. The susceptibility of these early T53-L-13 engines to nozzle assembly failure is detrimental to flight safety; therefore, they should be removed from service.

Oil Consumption

74. In addition to the engine failure, excessive oil consumption occurred with engine S/N LE14001. Oil consumption was 1 quart every 7 hours until approximately 75 hours of engine operation had elapsed. At that time, oil consumption suddenly increased to 4 quarts per hour. On teardown, it was found that a section of the forward element (approximately 2 inches long) of the number two bearing forward carbon seal (P/N 1-300-375-01) had broken off. Heavy oil coking was apparent on the air side of this seal and on the adjacent diffuser and compressor faces. Tailpipe streaking caused by this oil leak is shown in photo 8.

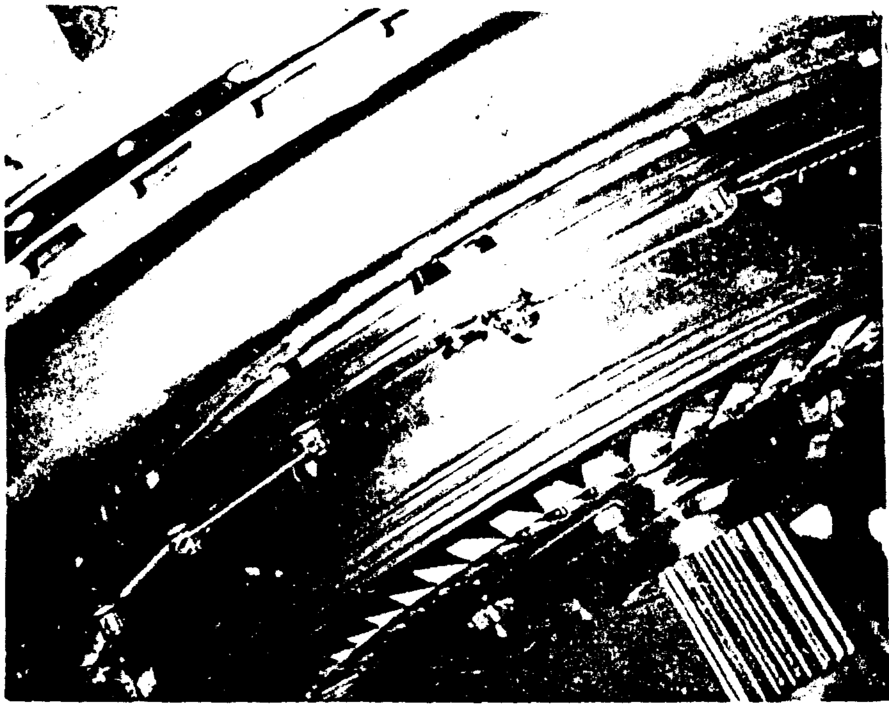

Photo 6. T53-L-13 N_2 Nozzle Crack S/N LE 14008.

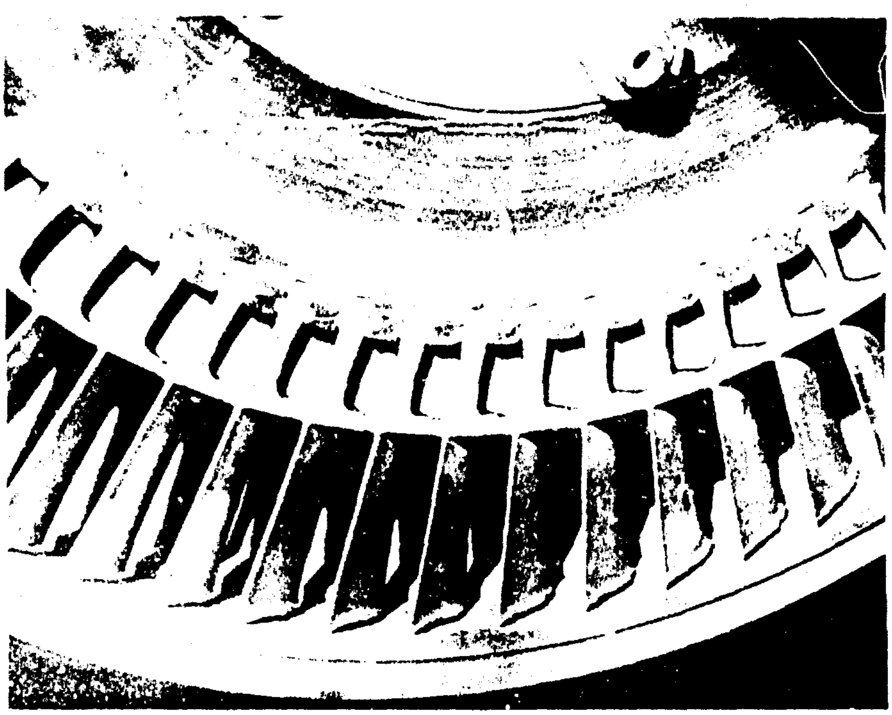

Photo 7. T53-L-13 Turbine Blade Tip Rubs.

Vibration

75. The vibration characteristics of the UH-1H were qualitatively evaluated during the level-flight performance tests. At gross weights of 6000 pounds and higher, the UH-1H's maximum airspeed was vibration limited and not V_{NE} limited. Vibration levels increased with altitude and were irritating at all weights and airspeeds at altitudes above 75 percent of service ceiling. At gross weights above 8500 pounds and at a rotor speed of 324 rpm, the vibration level became objectionable 5 to 10 knots below the published V_{NE} (ref 12, app I). Vibration levels increased as rotor speed decreased below 324 rotor rpm. The qualitative evaluation indicated a two-per-revolution vibration of high magnitude combined with random one-per-revolution lateral vibrations. For some conditions, below V_{NE}, vibration was severe enough to cause high cyclic feedback forces through the boost system and make the instruments unreadable. The most significant vibration effects occurred with combinations of high C_T, high forward airspeed and low ambient temperature. At these conditions, the Mach number of the rotor tip exceeded the critical Mach number of the blade. The compressibility combined with retreating blade stall produced unacceptable vibration characteristics at the published V_{NE}. Operation at high angles of attack and high tip Mach numbers results in a center of pressure shift which causes blade feathering and torsion loads. A revised V_{NE} schedule based on observed vibration is presented in figure 131, appendix II.

Main Rotor Shutdown

76. In winds above 20 knots, main rotor windmilling greatly complicated shutdown. It took 12 minutes for the main rotor to stop in winds of 20 knots gusting to 25 knots. The rotor maintained a constant speed of 25 rpm in winds of 30 knots. This violates paragraph 3.5.1 of MIL-H-8501A (ref 7, app I). A rotor deceleration device to be used only under adverse conditions would contribute to personnel safety and equipment protection.

Engine Oil Pressure Indicator

77. The engine oil pressure indicator is unsatisfactory. The operating oil pressure of the T53-L-13 engine is 90 (±10) psi. The maximum scale of present indicator is 100 psi. The pressure indication for the engines used during the test program was 100 psi at maximum power. This indication was at the maximum range of the indicator; therefore, it was impossible for the pilot to determine if the oil pressure actually exceeded maximum limits. It is mandatory that an oil pressure indicator with an expanded scale be installed in the UH-1H helicopter.

Fuel Capacity and Location

78. The operator's manual (TM 55-1520-210-10) indicates that total usable fuel (overfill) capacity is 224 gallons (ref 12, app I). The test aircraft had a total fuel capacity of 212 gallons when filled to filler lip which included 1.4 gallons of unusable fuel. Since this was a prototype aircraft, a production model (S/N 65-9689) was checked. It was found to have a total capacity of 217 gallons including 1.4 gallons of unusable fuel. In addition, the fuel cg of both aircraft was found to be significantly forward of that stated in the operator's manual. These conditions are nonconservative and could contribute to the aircraft being loaded out of cg limits and/or the lack of sufficient fuel for maximum range missions.

Weight and Balance

79. Weight and balance of the test aircraft is presented in appendix IV. Since this was a prototype aircraft and was configured for test purposes, it was not used to determine the weight and balance of a normally configured aircraft. Three methods were used to determine the weight and balance of a normal UH-1H:

a. A production UH-1D (S/N 65-9687) was weighed. Installed were armored seats and all presently used avionics. Fuel tanks were drained, but all other fluid reservoirs were left in their normal service condition. The results of the weighing are: weight, 5355 pounds and longitudinal cg location, FS 145.24. To correct this weight and balance to the UH-1H configuration, the following changes were made: addition of 11 pounds at station 166 for the particle separator and addition of 34 pounds at station 187 for the T53-L-13 engine conversion. This resulted in a corrected weight and balance of 5400 pounds at a longitudinal cg location of 145.55.

b. Weight and balance records of UH-1H, S/N 66-16386), being prepared for shipment overseas, were obtained from Sharpe Army Depot. These records indicated a weight and balance of 5265 pounds at a longitudinal cg location of 146.9. It is interesting to note that this aircraft had never been weighed. The weight and balance data for this helicopter were based on another aircraft with numerous items added and deleted from its weight records.

c. The following items and their weights were added to the airframe model specification empty weight of 4973 pounds: armored seat installation, 211 pounds; cargo mirror, 6 pounds; trapped fuel, 9 pounds; engine oil, 28 pounds; troop seats, 32 pounds; troop commander seats, 24 pounds; litter attendant seat, 3 pounds; safety belts, 11 pounds; stanchions, 14 pounds. This resulted in a total weight of 5351 pounds.

The weight and balance of the aircraft that was actually weighed (method a) is considered to be the most valid. The differences in weight determined by the other methods may be due to variance of installed avionics or some other items not considered. Because of the effect of weight and balance on the control problems encountered during these tests (paras 47 through 55), each aircraft should be weighed prior to operational use so an accurate weight and balance can be determined.

Operational Loadings

80. Because of the significant effects of aircraft weight and balance on the control problems experienced during low-speed flight, various operational loads were investigated. The basic aircraft weight was 5400 pounds with a longitudinal cg location at station 145.55 as determined in paragraph 79. The weight and longitudinal cg with fuel burn-off for various troop loadings are shown in figure AD (unarmed) and in figure AE (armed with two XM23 machine guns and 2400 rounds of ammunition). The following assumptions were made: troop weight, 220 pounds each; armament weight, 123 pounds at station 142.3 (guns and mounts); ammunition, 156 pounds at station 140.4; and fuel, 6.5 pounds per gallon. Without armament, fuel must be limited to the following amounts to remain within the present weight/cg envelope: with 12 troops, 480 pounds; with 11 troops, 1240 pounds; with less than 11 troops, no limit. To remain within the recommended restricted weight/cg envelope, fuel must be limited as follows: with 12 troops, not possible; with 11 troops, 100 pounds; with 10 troops, 600 pounds; with 9 troops, 1430 pounds (full fuel). With armament, fuel must be limited to the following amounts to remain within the present weight/cg envelope: with 12 troops, 280 pounds; with 11 troops, 960 pounds; with 10 troops, 1180 pounds; with 9 troops, 1400 pounds; with 8 troops, 1430 pounds (full fuel). To remain within the recommended restricted weight/cg envelope with armament, fuel must be limited as follows: with 11 or 12 troops, not possible; with 10 troops, 450 pounds; with 9 troops 1430 pounds (full fuel). If pilot and copilot are always carried, the basic aircraft cg could be moved aft by adding tail boom ballast or by changing the location of the avionics. This would allow significantly more fuel or more troops to be carried while remaining within cg limits. With the internal ferry tanks installed and one 170-pound pilot, the aft cg limit can be easily exceeded. This loading and fuel burn-off weight and cg are shown in figure AF. Pilots should be cautioned to move the battery forward and/or to load cargo forward to avoid exceeding the aft cg limit when in the ferry configuration or with the armored seats removed. A caution notice could be stenciled on the internal ferry tanks.

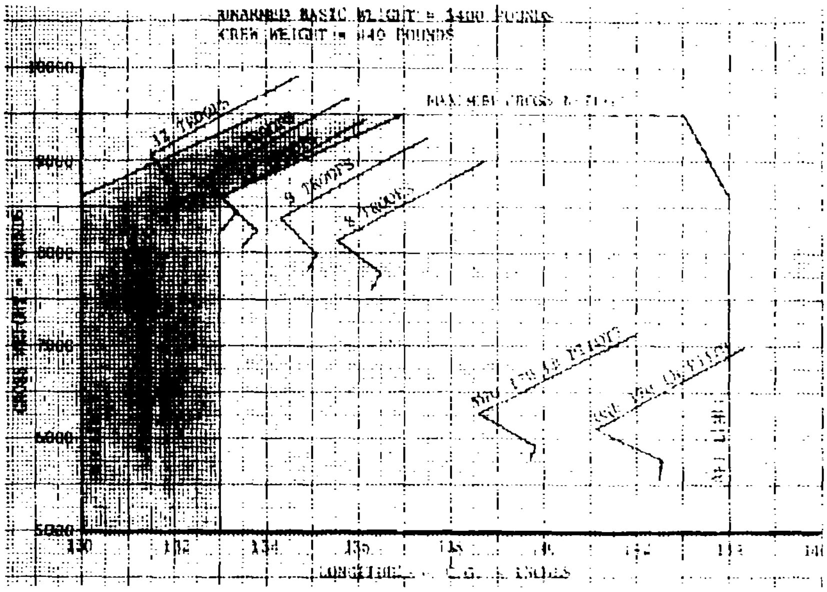

Figure AD. Weight and CG for Troop Loading, Fuel Burnoff Lines.

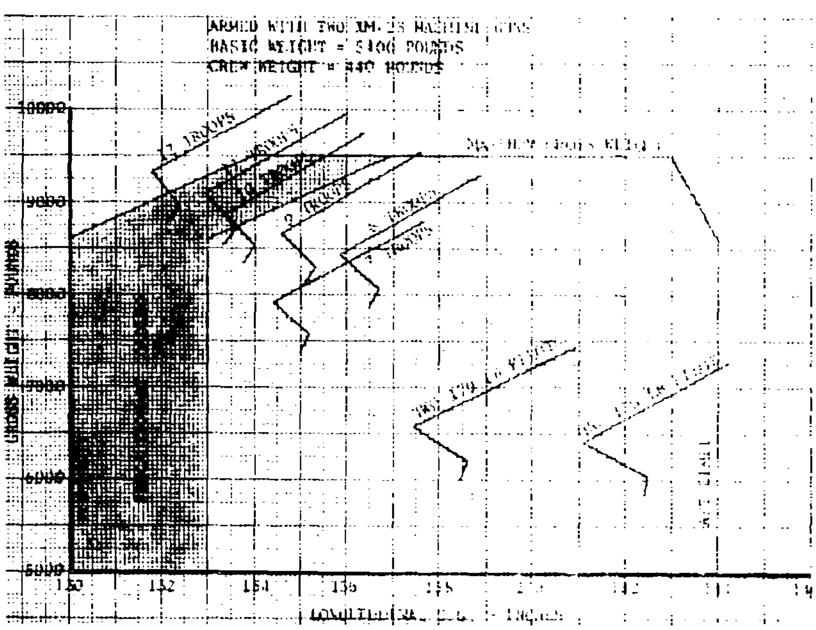

Figure AE. Weight and CG for Troop Loading, Fuel Burnoff Lines.

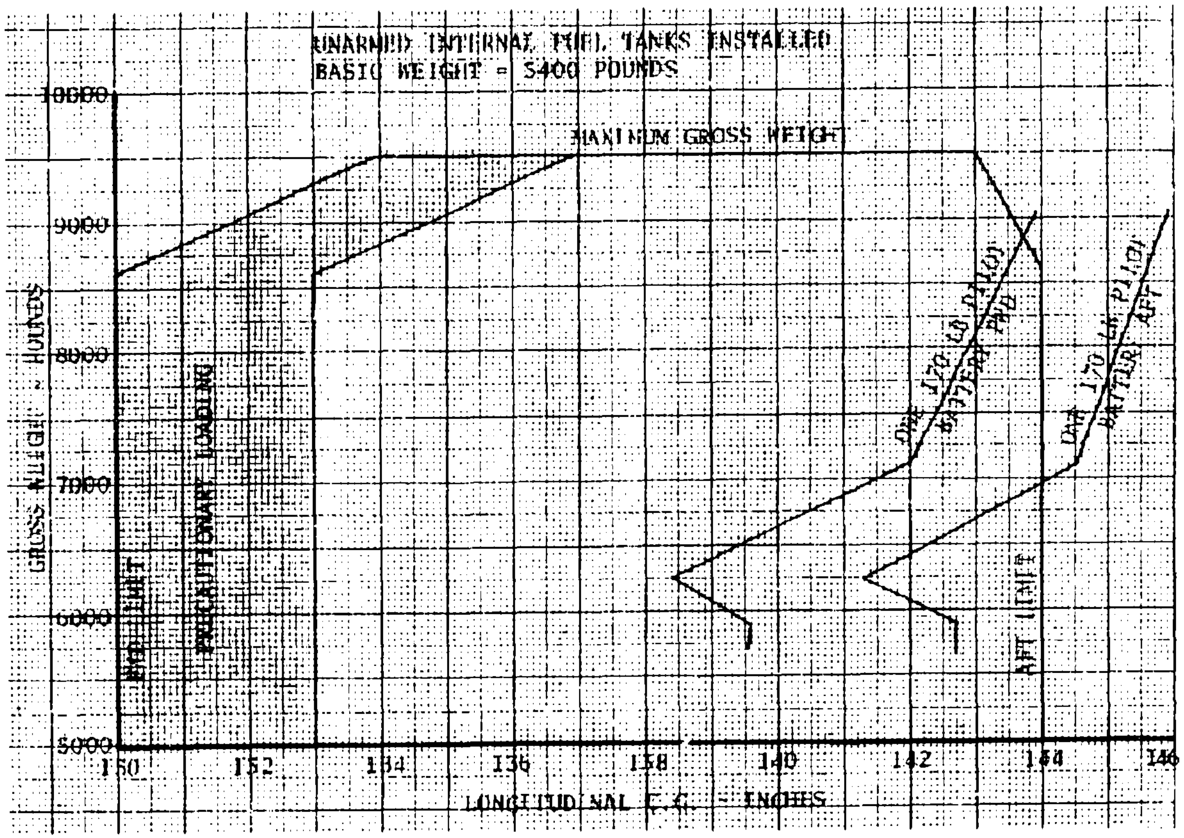

Figure AF. Weight and CG for Ferry Loading Fuel Burnoff Lines.

CONCLUSIONS

GENERAL

81. The following general conclusions were reached as a result of the limited Phase D test of the YUH-1H helicopter:

a. Hovering performance of the YUH-1H as compared to that of the UH-1D is improved by approximately 1000 pounds for conditions where directional control, torque or weight limitations are not encountered (para 10).

b. Takeoff performance of the YUH-1H at a given altitude is comparable to that of the UH-1D at a 5000-foot lower altitude (para 16).

c. Climb performance as compared to the UH-1D is substantially improved. Service ceilings of the YUH-1H are approximately 3000 feet higher than those of the UH-1D (para 22).

d. Range performance as compared to the UH-1D is improved from 1 to 10 percent (para 38).

e. The acceleration characteristics of the T53-L-13 engine are substantially better than those of earlier model T53 engines (para 70).

f. The UH-1H cannot be safely operated throughout the current envelope due to inadequate longitudinal and directional control (para 54).

g. The left pedal control margin is adversely affected by increased power required, gross weight and density altitude and by decreased rotor speed. Directional handling qualitites were also adversely affected as the longitudinal cg moved forward (para 50).

h. The aft longitudinal control margin was adversely affected by increased gross weight and forward cg location (para 51).

DEFICIENCIES AND SHORTCOMINGS

82. Correction of the following deficiencies is mandatory for acceptance of the aircraft:

a. Insufficient longitudinal and directional control for safe operation of the aircraft within the present low-speed flight envelope (para 55).

b. Susceptibility to failure of the T53-L-13 engine equipped with the machined N_1 nozzle (para 73).

83. Correction of the following shortcomings is desirable for inproved operation and mission capability:

a. Undesirable discontinuity in pedal effectiveness (para 50).

b. Excessive tail-low attitude when landing with a cg aft of station 140 (para 53).

c. Ease with which UH-1H torque limits may inadvertently be exceeded during takeoffs or other high-powered operations (para 15).

d. Ambiguous and erroneous power, torque and speed limits as shown in the various UH-1H manuals (para 56).

e. Poor torque system accuracy and poor torquemeter readability (para 56).

f. Unsatisfactory power turbine speed control (droop compensation) during power changes (para 71).

g. Undesirable engine speed oscillation when operating at maximum N_1 (topping) power (para 70).

h. Poor climb and level flight dynamic stability characteristics (para 45).

i. Vibration limited level flight airspeeds and range performance at high gross weight and/or altitude (paras 37 and 75).

j. Poor static lateral-directional stability characteristics (para 46).

k. Excessive airspeed system position error (para 43).

l. Rotor windmilling after engine shutdown in winds of 20 knots or more (para 76).

m. Incorrect UH-1D and UH-1H fuel capacity and cg location stated in the operator's manual (para 78).

n. Invalid method of ground trimming the N_1 governor of the T53-L-13 engine stated in the maintenance manual (para 72).

o. Unnecessarily high engine start exhaust gas temperatures due to use of procedures as stated in the operator's manual (para 69).

p. Difficulty in determination of ground idle/start position on throttle (para 69).

q. Inadequate range of engine oil pressure indicator (para 77).

r. Excessive maintenance manhours required to clean the air inlet filters (para 3, app VI).

RECOMMENDATIONS

84. The deficiencies, correction of which is mandatory, should be corrected prior to operational use of the UH-1H system (para 82).

85. The shortcomings, correction of which is desirable, should be corrected on a high-priority basis (para 83).

86. Incorporate the recommended precautionary loading envelopes as shown in figure 130 into the operator's manual (para 55).

87. Incorporate into the operator's manual the method of recovering from loss of control caused by insufficient left pedal or aft longitudinal control (paras 50 and 51).

88. Revise all indicator limitations as presented in the cockpit (red lines, V_{NE}, etc.) to correct indicated values (paras 43 and 56).

89. Install an overtorque warning light in the UH-1H helicopter (para 15).

90. Investigate the feasibility of incorporating an automatic torque limiter with an override feature (para 15).

91. Incorporate the following warnings in the UH-1H operator's manual:

a. Weight and balance limitations must be rigidly observed as the UH-1H is control and structurally limited and generally not power limited.

b. There is insufficient left directional control during hover, takeoff or landing in adverse winds at 9500 pounds gross weight at 5000 feet, and lower weights at higher altitudes (para 50).

c. Inadequate aft longitudinal control exists when longitudinal cg is within 3 inches of the present forward limit (para 51).

d. As cg moves forward, gross weight should be reduced to insure adequate left pedal margin. At full forward cg, the gross weight limit for adequate directional control is 1200 pounds less than that shown on figure 130 (para 50).

e. Torque indications must be closely monitored as the primary power instruments below the engine critical altitude (para 15).

92. Incorporate the following caution notes in the operator's manual:

a. When flying at an aft cg (station 140 to 144), terminate an approach at a minimum 5-foot hover prior to landing to prevent striking the tail on the ground (para 53).

b. With the armored seats installed and a left-lateral cg, the pilot's arm and right cyclic movement will be restricted (para 49).

c. Takeoffs and landings should be limited to those conditions at which the UH-1H can hover at 5 feet as shown in figure 2 (paras 18 and 21).

93. To provide the pilot with additional information, make additions or changes to the operator's manual as follows:

a. Revise the performance data to reflect the results of this report.

b. Engine and rotor speed oscillations may be prevented by reducing power ½ to 1 percent below N_1 topping power (para 70).

c. When operating with the internal ferry fuel tanks or with armored seats removed, the battery should be moved to the forward location and/or cargo loaded forward to remain within cg limits[1] (para 80).

d. Add a note explaining the better method of determining maximum continuous power setting (para 56).

e. Revise the operator's manual to show the correct usable fuel capacity (para 78).

f. When operating at low airspeeds (takeoff, landing, hover, taxi, etc.) use 324 rotor rpm (6600 engine rpm) (paras 10, 50, and 52).

g. Steep or vertical takeoffs and landings should be limited to those conditions at which the UH-1H can hover OGE as shown in figure 1, appendix II (paras 18 and 21).

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[1]This could be stenciled on ferry tanks (para 80).

94. An actual weighing of each helicopter should be conducted prior to its operational use and checked periodically (para 79).

95. Air inlet filters should be designed to minimize or eliminate cleaning requirements (para 3, app VI).

96. Conduct quantitative performance, stability and control, stress and vibration tests on the final production configuration UH-1H helicopter throughout the operational temperature range (paras 7, 11, 30, 44, 49, and 55).

APPENDIX I. REFERENCES

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17. Specification, Bell Helicopter Company, No. 205-947-065, *Detail Specification for UH-1D Utility Helicopter FY 68 Procurement,* 15 March 1967.

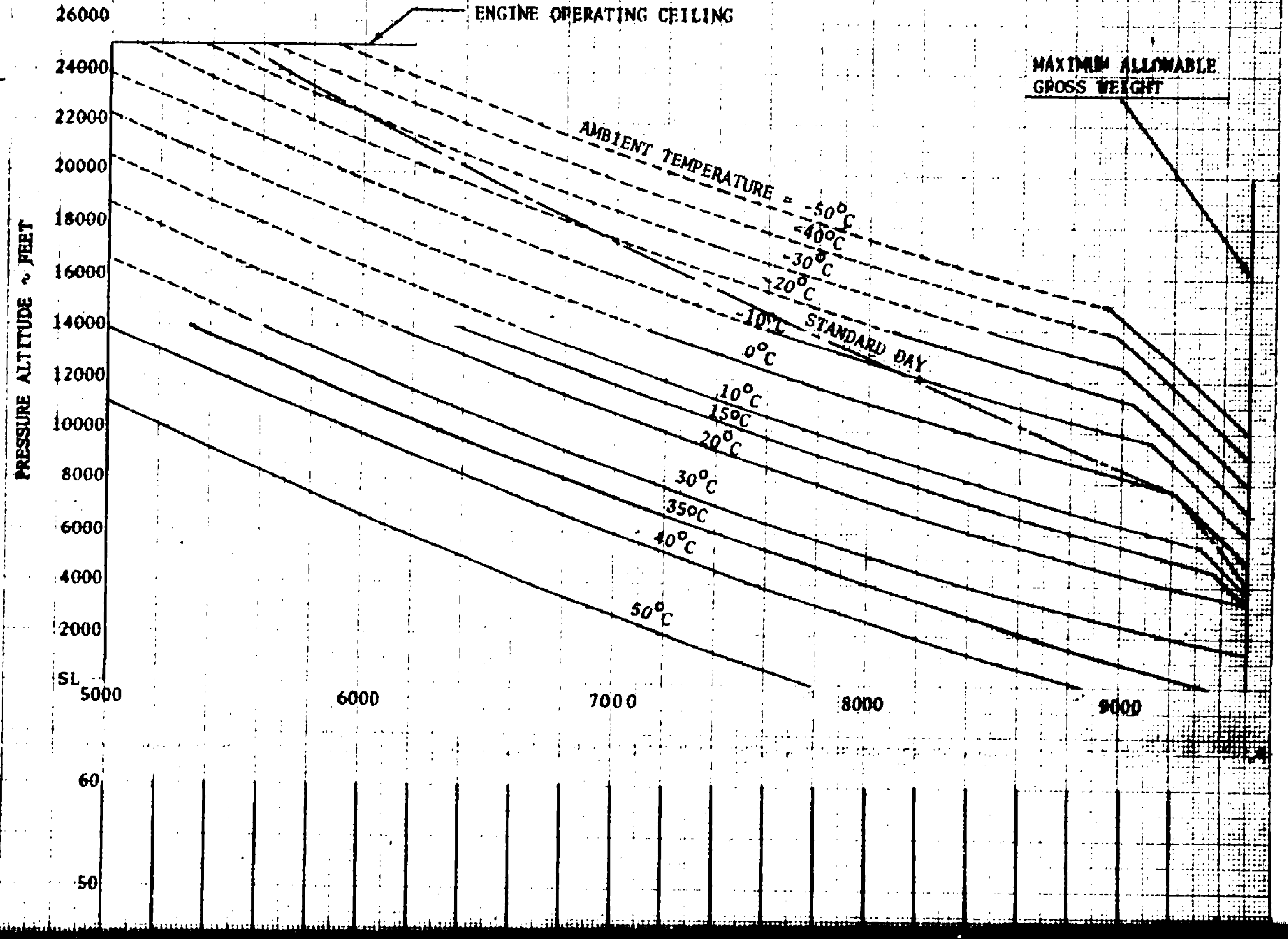
18. Final Report, USAASTA, Project 66-06, Engineering Flight Test of the AH-1G Helicopter, Phase D, Part 1, to be published.

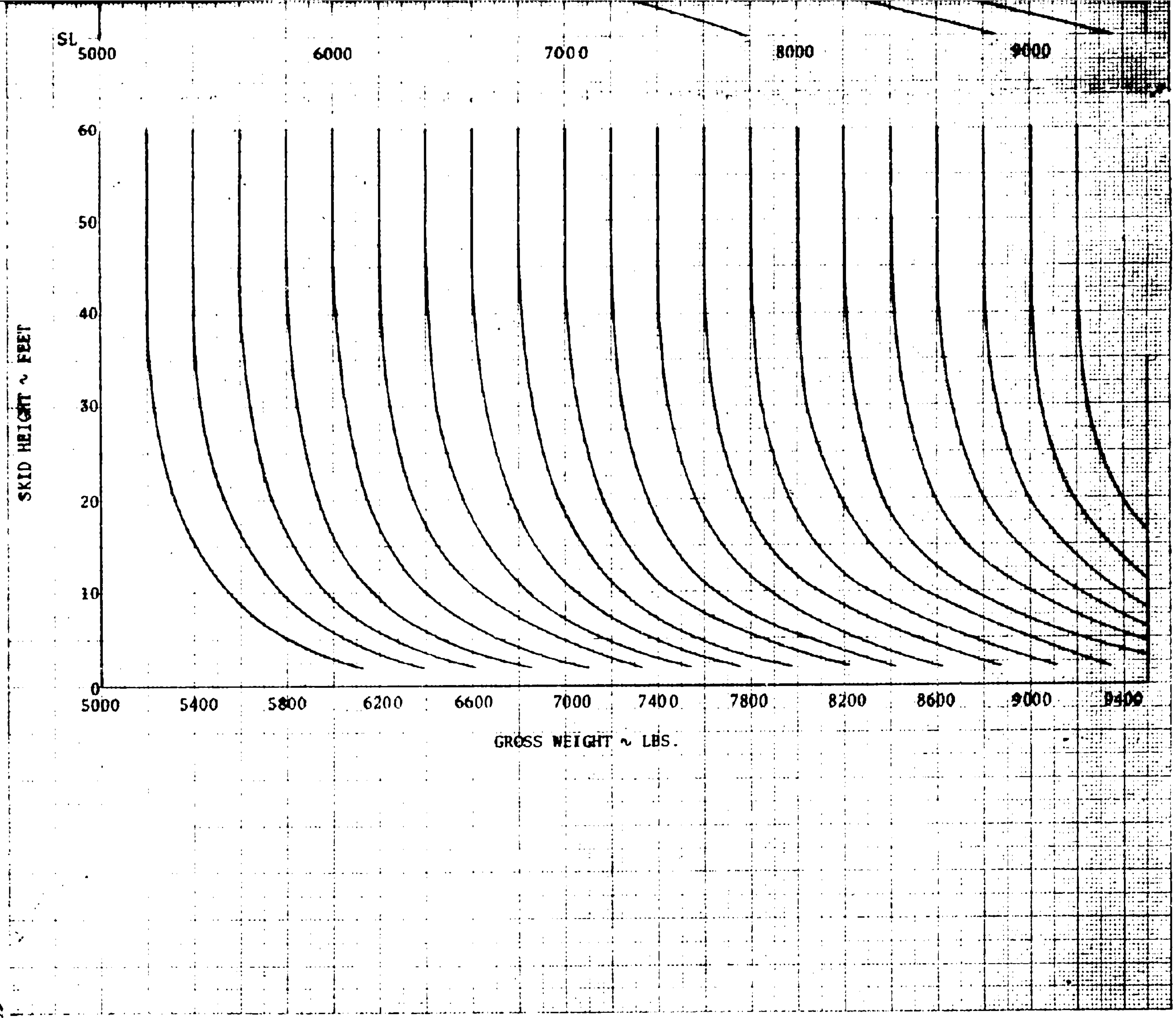
APPENDIX II. TEST DATA

INDEX

FIGURE 1
SUMMARY HOVERING PERFORMANCE
UH-1H T53-L-13
MILITARY POWER 324 ROTOR RPM

NOTE: 1. Derived From Figures 6 And 116
2. This performance is based on maximum power available and cannot be achieved at most conditions because of insufficient directional control (see fig 130)





SL
5000
6000
7000
8000
9000
SKID HEIGHT ~ FEET
60
50
40
30
20
10
0
5000
5400
5800
6200
6600
7000
7400
7800
8200
8600
9000
9400
GROSS WEIGHT ~ LBS.

11

FIGURE 2

IN GROUND EFFECT HOVERING CEILING AT 5 FOOT SKID HEIGHT
(RECOMMENDED MAXIMUM LANDING CEILING)
UH-1H T53-L-13
MILITARY POWER 324 ROTOR RPM

NOTE: 1. Derived from Figures 8 and 116
2. This performance is based on maximum power available and cannot be achieved at most conditions because of insufficient directional control (see fig 130)

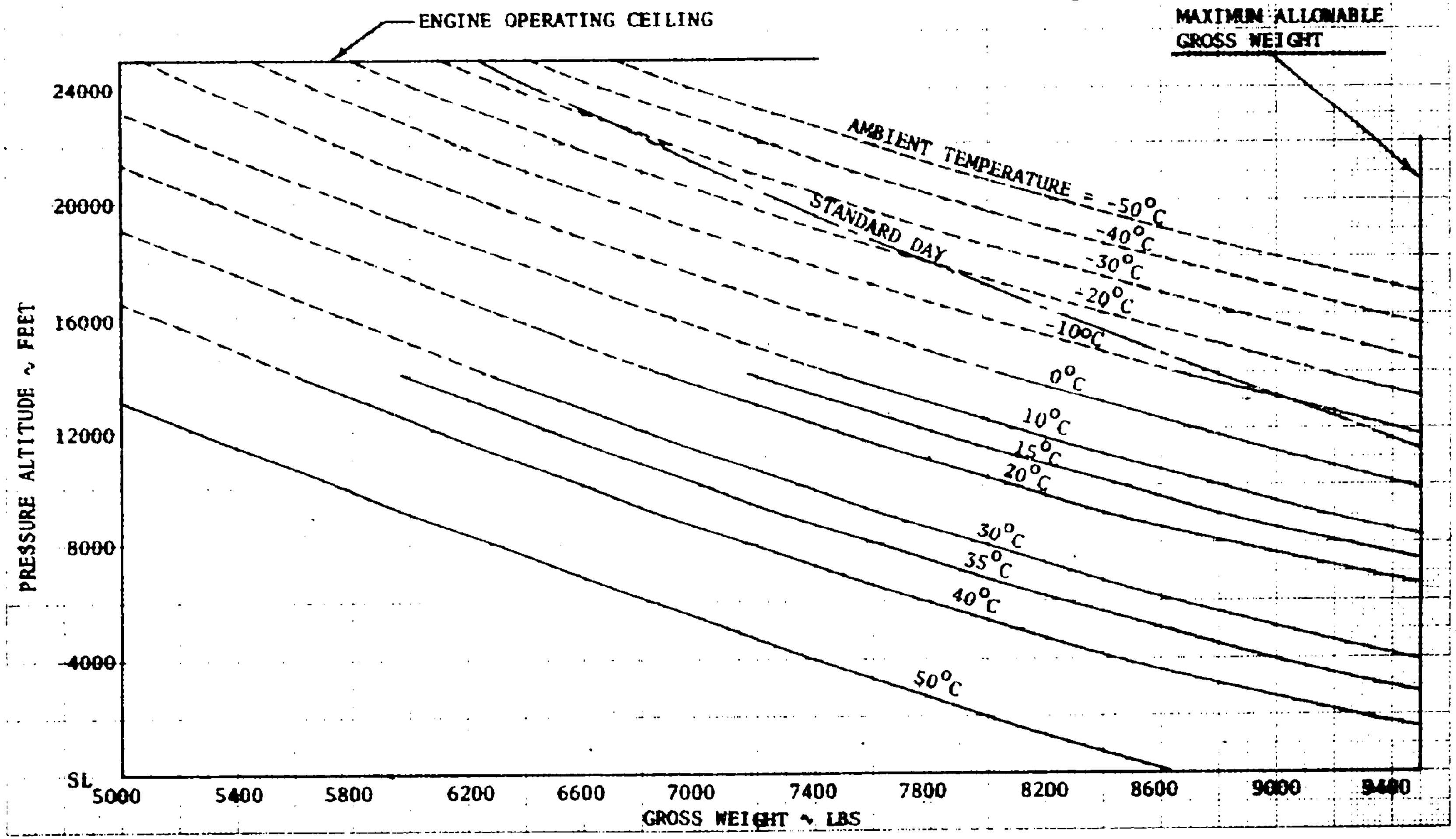


FIGURE 3

IN GROUND EFFECT HOVERING CEILING AT 2 FOOT SKID HEIGHT

(RECOMMENDED MAXIMUM TAKEOFF CEILING)

UH-1H T53-L-13

MILITARY POWER 324 ROTOR RPM

NOTE: 1. Derived from Figures 7 and 116

2. This performance is based on maximum power available and cannot be achieved at most conditions because of insufficient directional control (see fig 130)

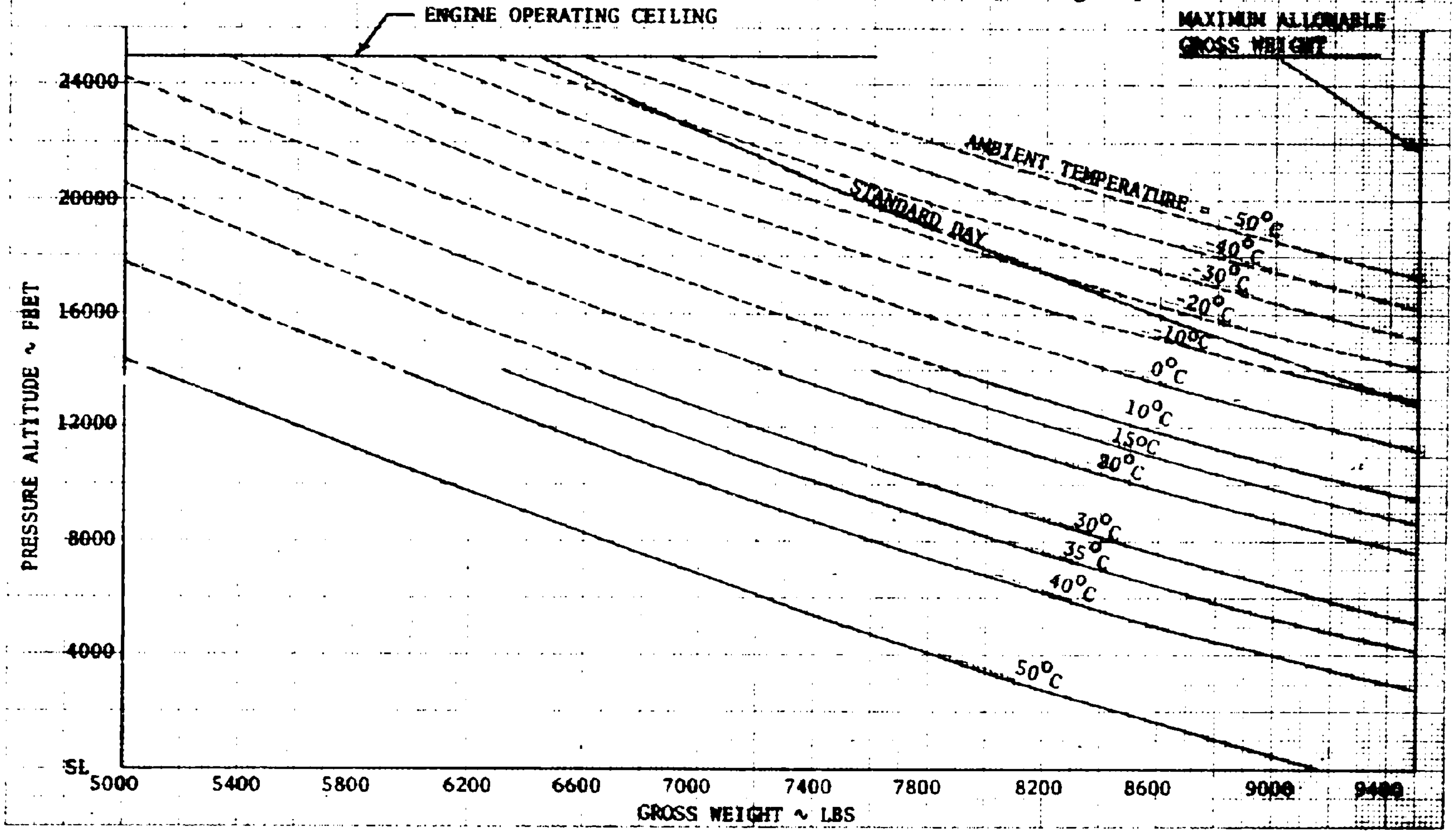


FIGURE 4
POWER REQUIRED TO HOVER OUT OF GROUND EFFECT
YUH-1H S/N 60-6029
324 ROTOR RPM

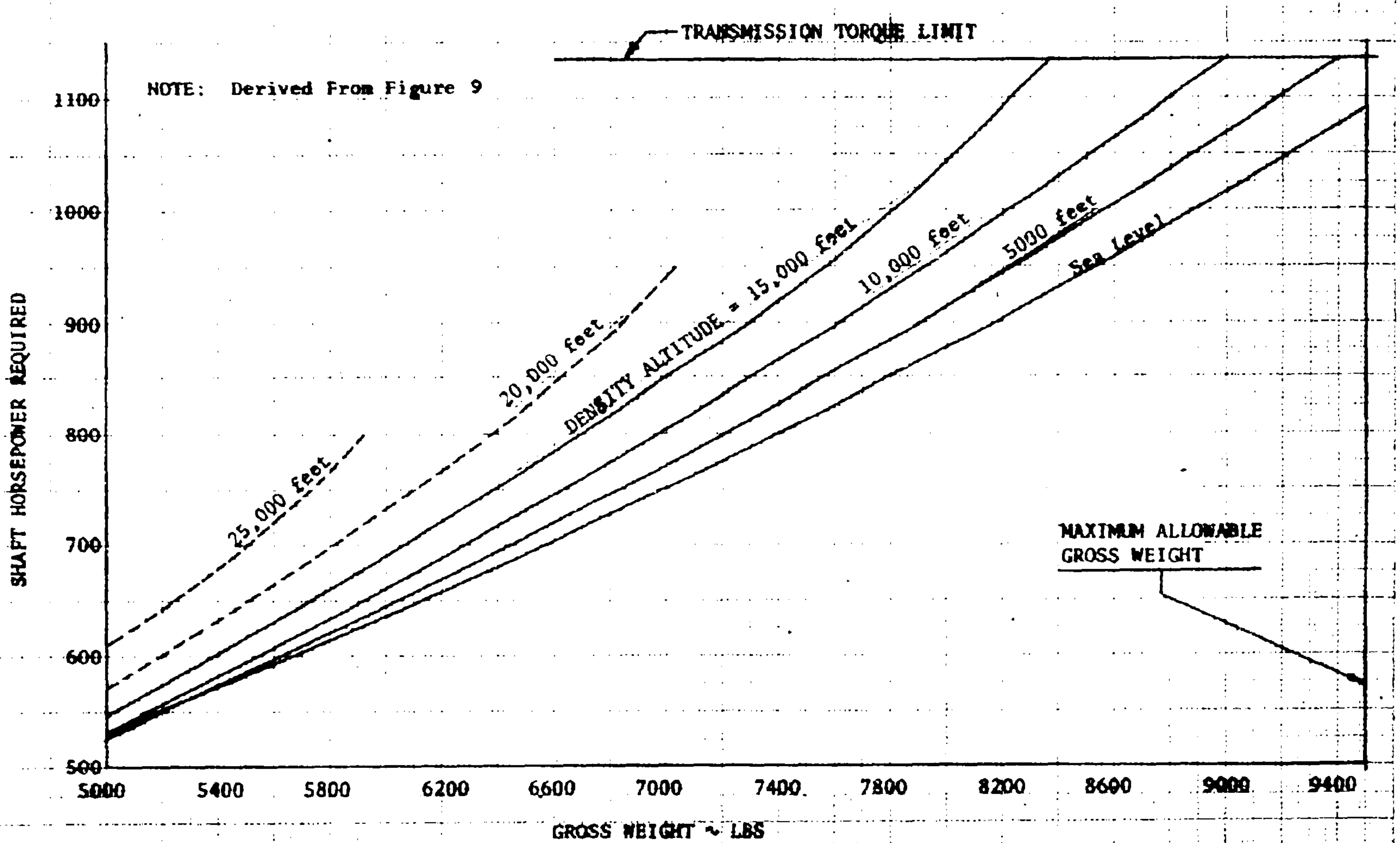


FIGURE 5
POWER REQUIRED TO HOVER AT 2 FOOT SKID HEIGHT
YUH-1H S/N 60-6029
324 ROTOR RPM

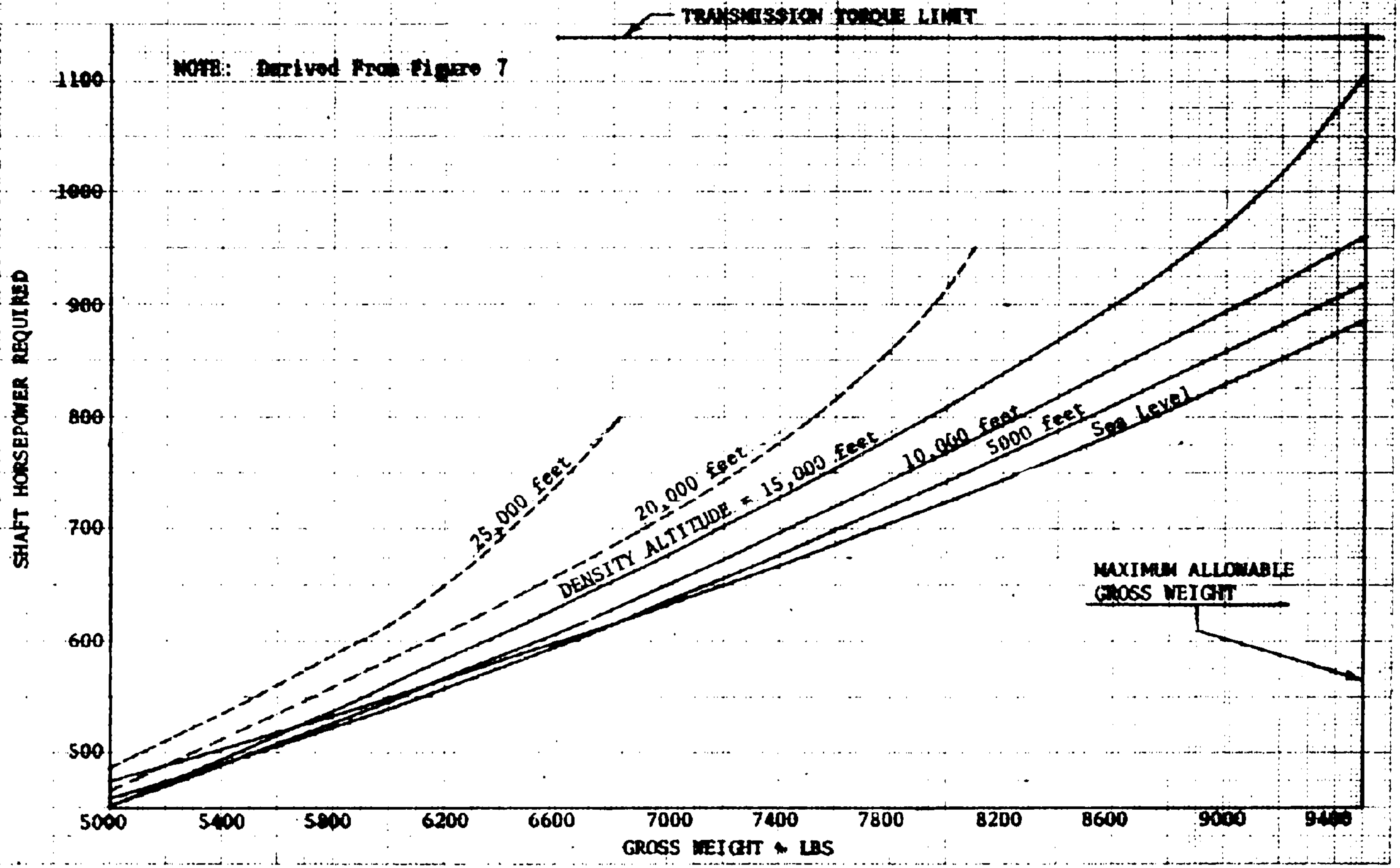


FIGURE 6
NONDIMENSIONAL HOVERING PERFORMANCE SUMMARY
YUH-1H S/N 60-6029

NOTES:
1. Distance from lowest part of skids to center of rotor hub = 12 feet 5 3/4 inches
2. Solid curves taken from Figures 7 through 9
3. Dashed curves obtained from Ref. 5
4. Data flown in less than two knots of wind

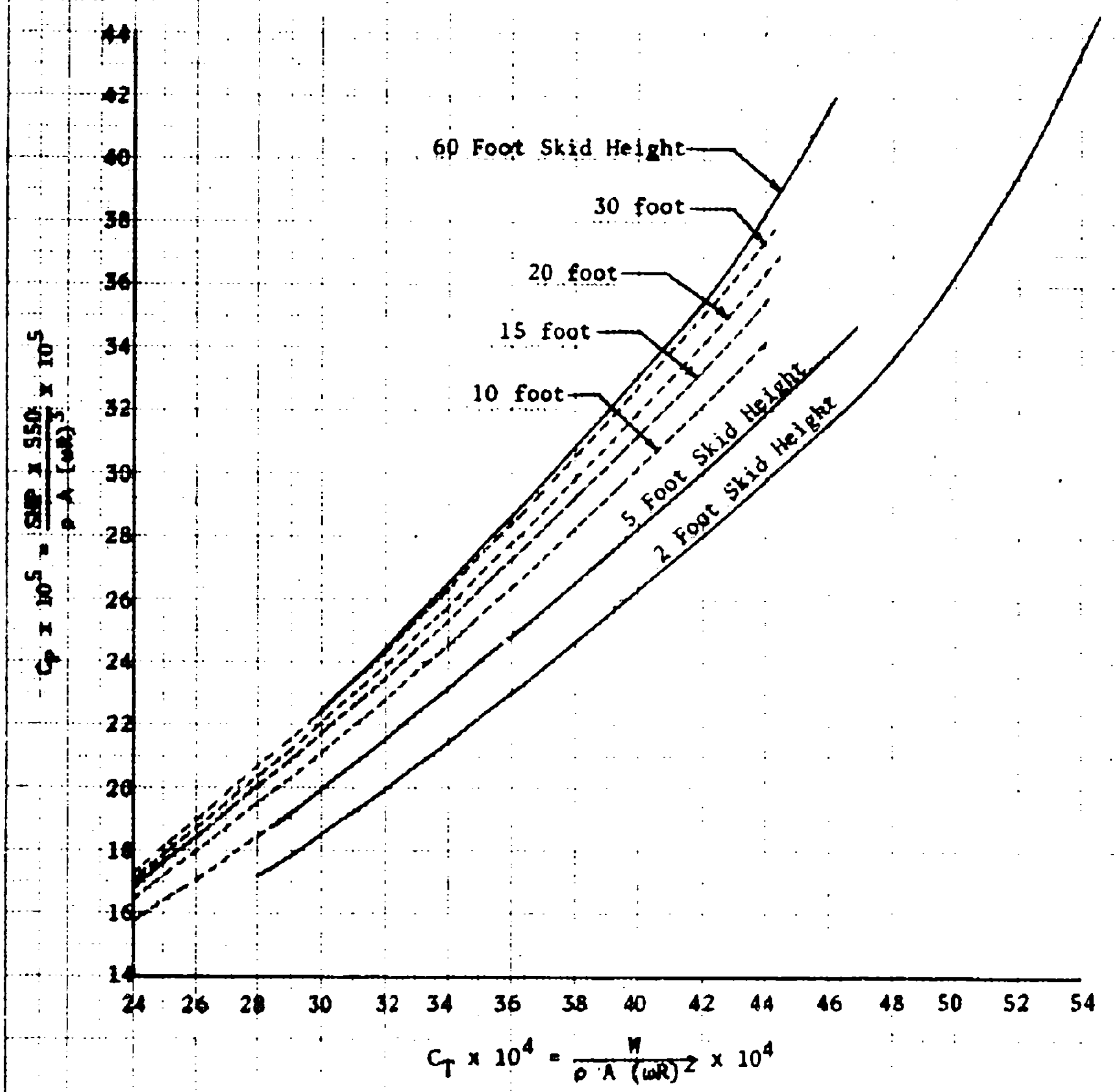


FIGURE 7
NON DIMENSIONAL HOVERING PERFORMANCE
YUH-1H S/N 60-6029
TETHERED FLIGHT
2 FOOT SKID HEIGHT

NOTES:

1. Distance from lowest part of skids to center of rotor hub = 12 feet 5 3/4 inches
2. Data flown in less than two knots of wind
3. Solid symbols denote free flight hover

ROTOR SPEED ~ RPM 324	314	DENSITY ALTITUDE ~ FEET	AMBIANT TEMP ~ °C
○		2800	3
	⌀	3670	3
	△	4050	5
	⌀	3750	2
◻	◻	5260	-16
◇		11970	- 7
	◇	11820	- 9
⬡	⬡	12240	2
☆	☆	12100	- 5
	⌀	12500	2

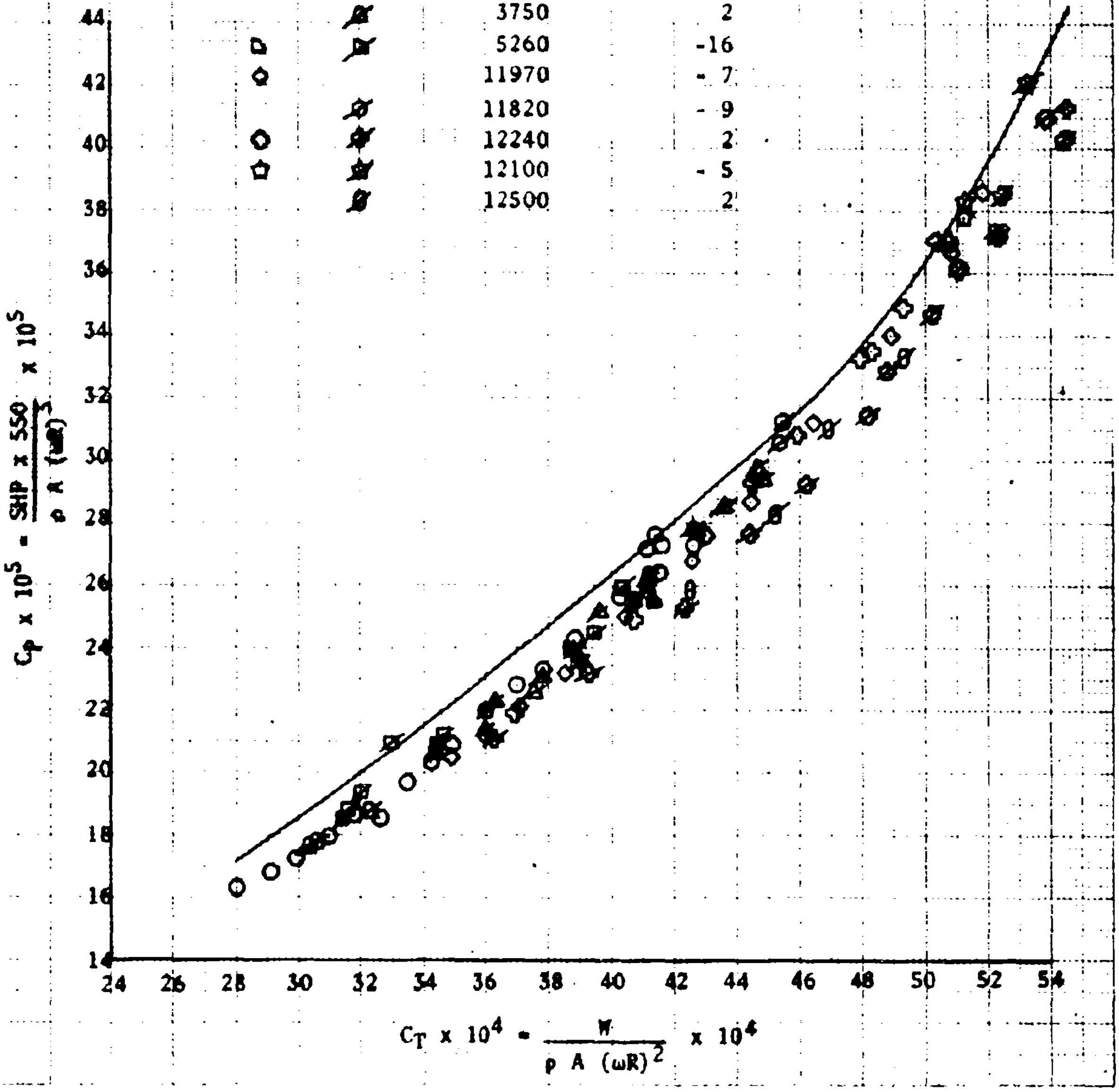


FIGURE 8
NON DIMENSIONAL HOVERING PERFORMANCE
YUH-1H S/N 60-6029
TETHERED FLIGHT
5 FOOT SKID HEIGHT

NOTES:

1. Distance from lowest part of skids to center of rotor hub = 12 feet 5 3/4 inches
2. Data flown in less than two knots of wind
3. Solid symbols denote free flight hover

ROTOR SPEED ~ RPM 324	314	DENSITY ALTITUDE ~ FEET	AMBIANT TEMP ~ °C
O		4790	11
◻	◻	5650	21

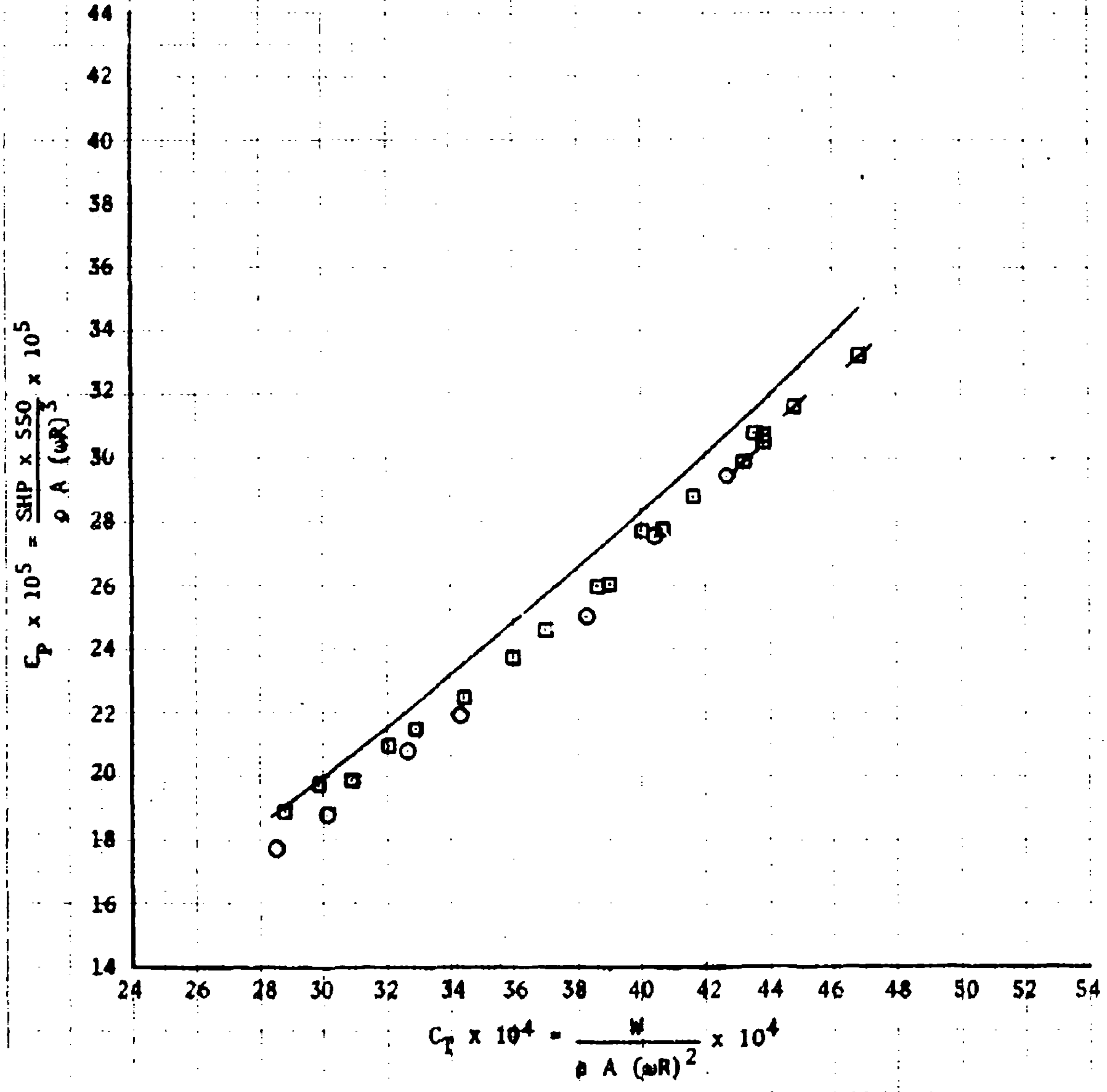


FIGURE 9
NON DIMENSIONAL HOVERING PERFORMANCE
YUH-1H S/N 60-6029
TETHERED FLIGHT
60 FOOT SKID HEIGHT

NOTES:

1. Distance from lowest part of skids to center of rotor hub = 12 feet 5 3/4 inches
2. Data flown in less than two knots of wind
3. Solid symbols denote free flight hover

ROTOR SPEED ~ RPM		DENSITY ALTITUDE ~ FEET	AMBIANT TEMP ~ °C
324	314		
○	⌀	4550	6
□		4600	12
△	⟁	12420	2

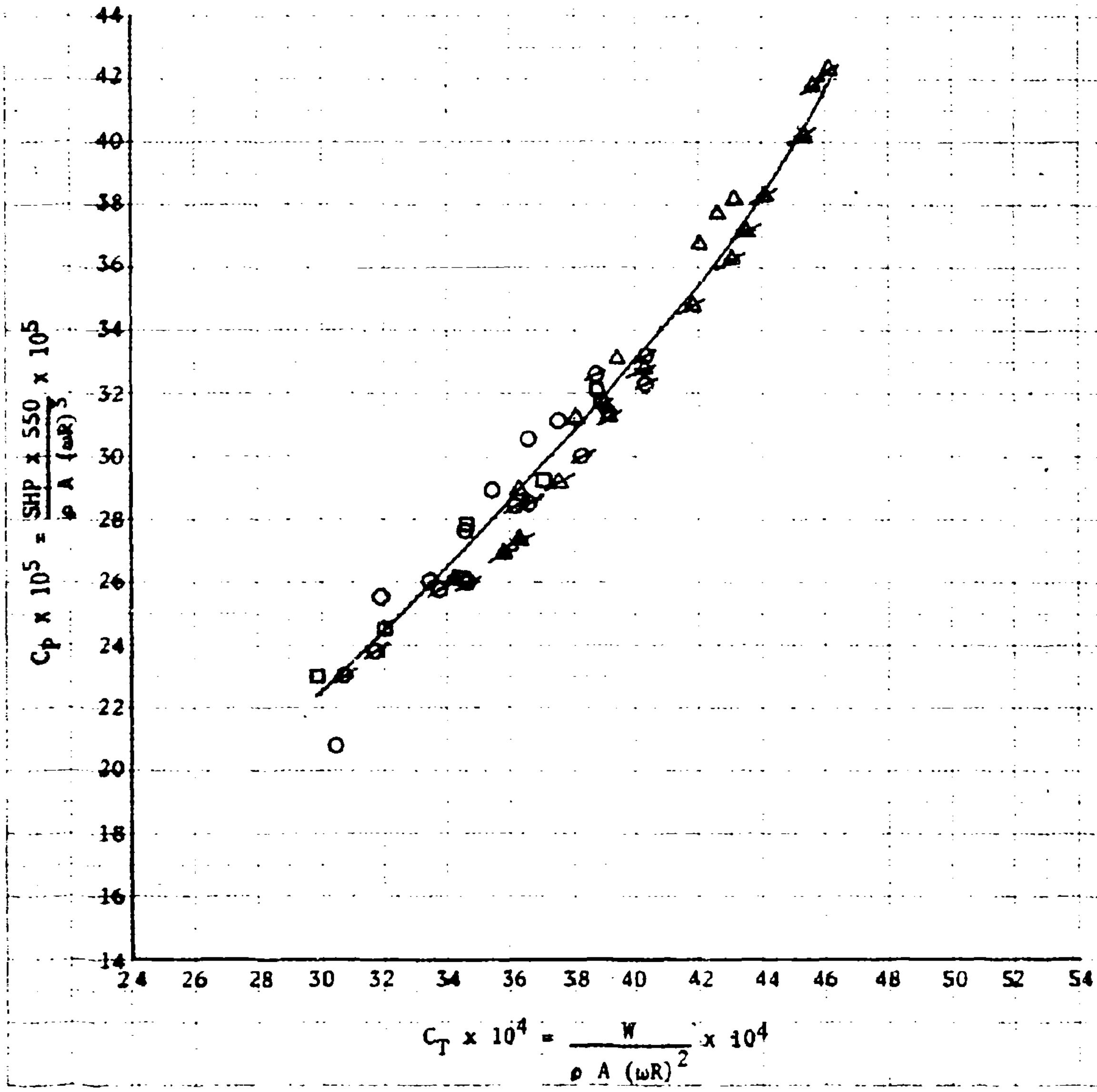


FIGURE 10
DIMENSIONAL TAKEOFF PERFORMANCE
UH-1H AND UH-1D

STANDARD DAY
GROSS WEIGHT = 9500 LB.
MILITARY POWER
ROTOR SPEED = 324 RPM

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER

NOTES: 1. Solid lines denote UH-1H derived from figures 7,9,16 and 116.
2. Dashed lines denote UH-1D derived from FTC-TDR-64-27 reference 5 figures 25 and 104

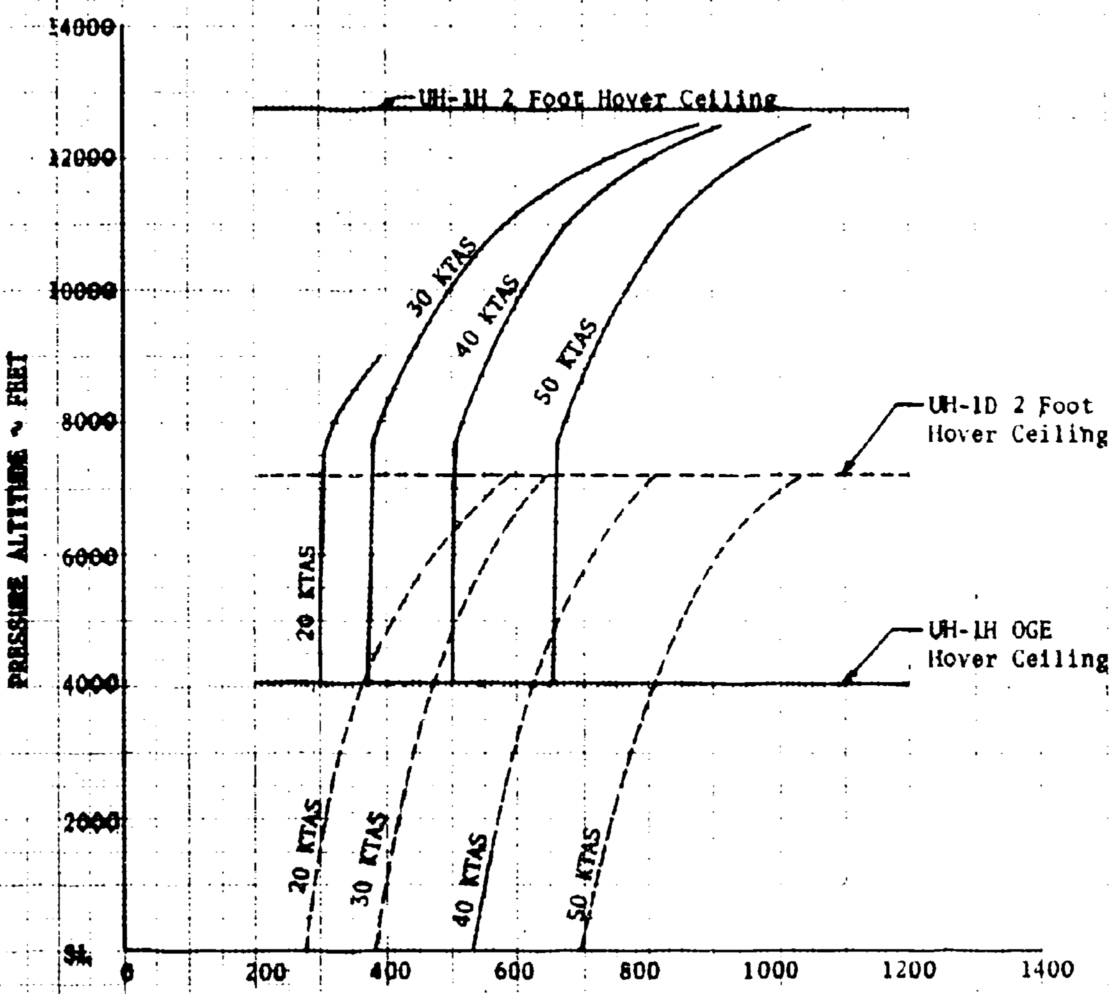


FIGURE 11
DIMENSIONAL TAKEOFF PERFORMANCE
UH-1H AND UH-1D

AMBIENT TEMPERATURE = 35°C
GROSS WEIGHT = 8500 LB.
MILITARY POWER
ROTOR SPEED = 324 RPM

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER

NOTES: 1. Solid lines denote UH-1H derived from figures 7,9,16 and 116

2. Dashed lines denote UH-1D derived from FTC-TDR-64-27 reference 5 figures 25 and 104

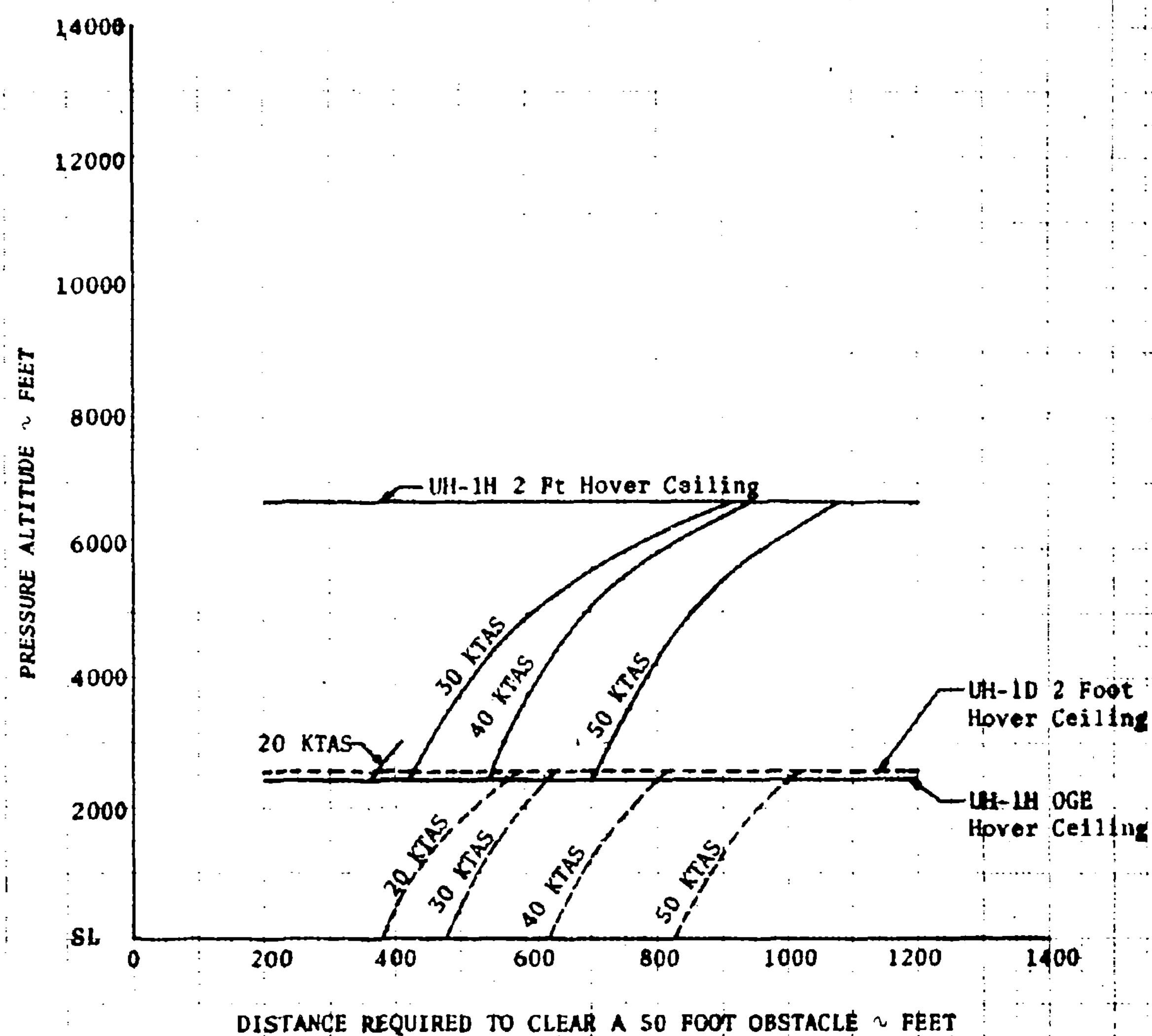


FIGURE 12
APPROXIMATE TAKEOFF PERFORMANCE
UH-1H T53-L-13

MILITARY POWER 324 ROTOR RPM
PRESSURE ALTITUDE = SEA LEVEL
TECHNIQUE LEVEL ACCELERATION FROM A 2 FOOT HOVER
WITH 30 KNOTS INDICATED AIRSPEED (35 KCAS) CLIMBOUT

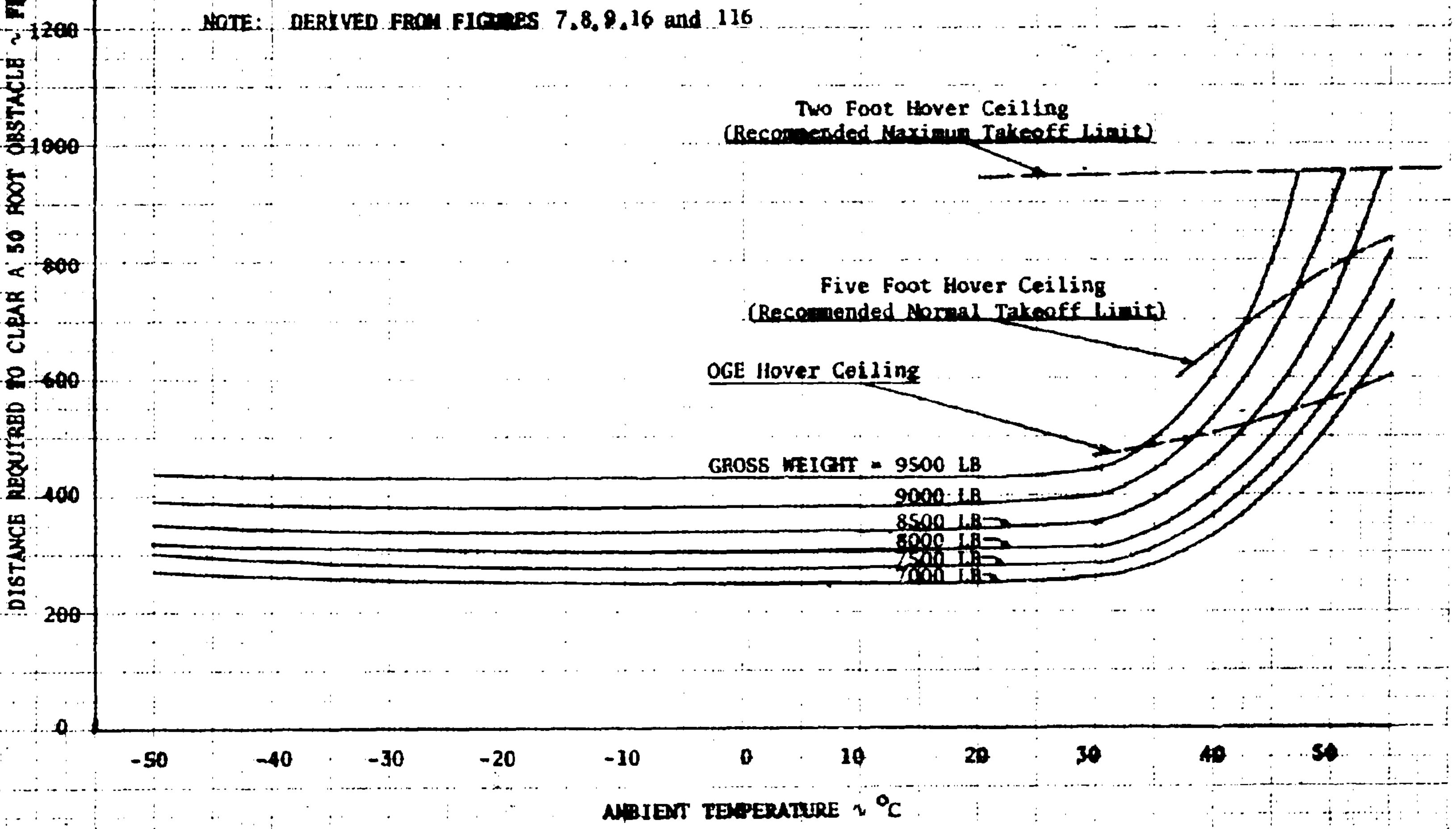


FIGURE 13

APPROXIMATE TAKEOFF PERFORMANCE

UH-1H T53-L-13

MILITARY POWER 324 ROTOR RPM

PRESSURE ALTITUDE = 5,000 FEET

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
WITH 30 KNOTS INDICATED AIRSPEED (35 KCAS) CLIMBOUT

NOTE: Derived from Figures 7,8,9,16, and 116

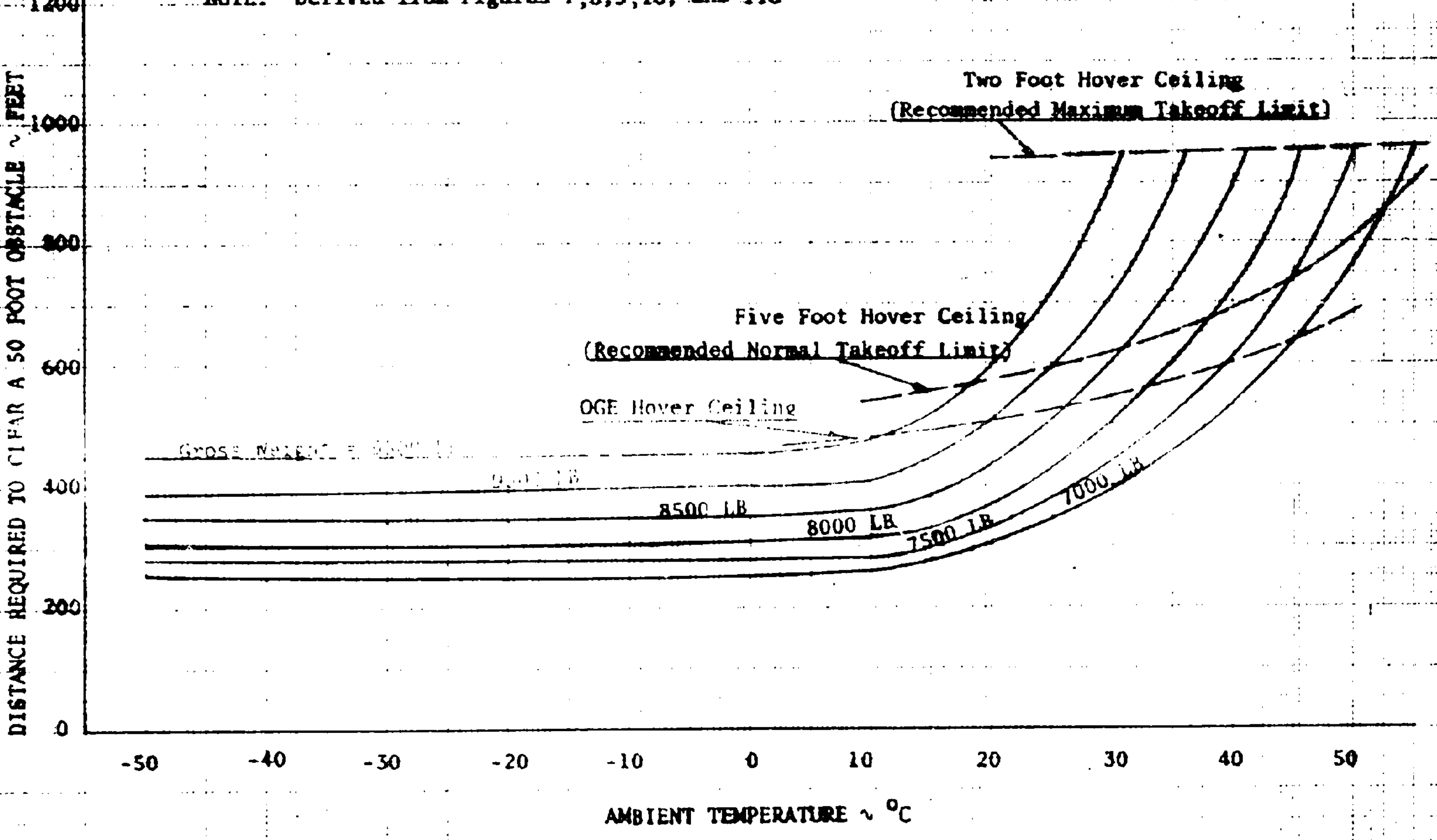


FIGURE 14
APPROXIMATE TAKEOFF PERFORMANCE

UH-1H T53-L-13

MILITARY POWER 324 ROTOR RPM

PRESSURE ALTITUDE = 10,000 FEET

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
WITH 30 KNOTS INDICATED AIRSPEED (35 KCAS) CLIMBOUT

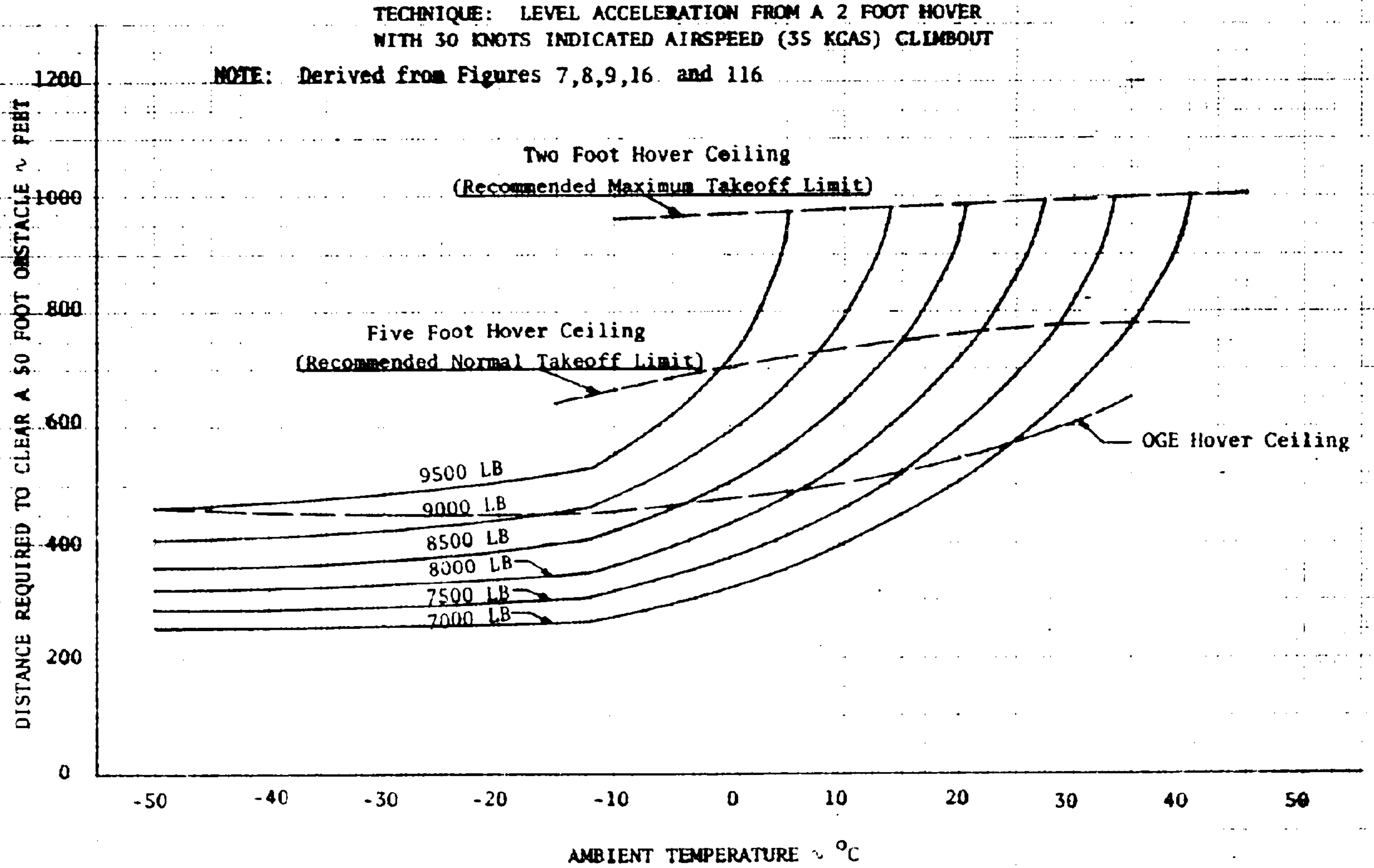


FIGURE 15

APPROXIMATE TAKEOFF PERFORMANCE

UH-1H T53-L-13

MILITARY POWER 324 ROTOR RPM

PRESSURE ALTITUDE = 15,000 FEET

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER WITH 30 KNOTS INDICATED AIRSPEED (35 KCAS) CLIMBOUT

NOTE: Derived from Figures 7,8,9,16, and 116

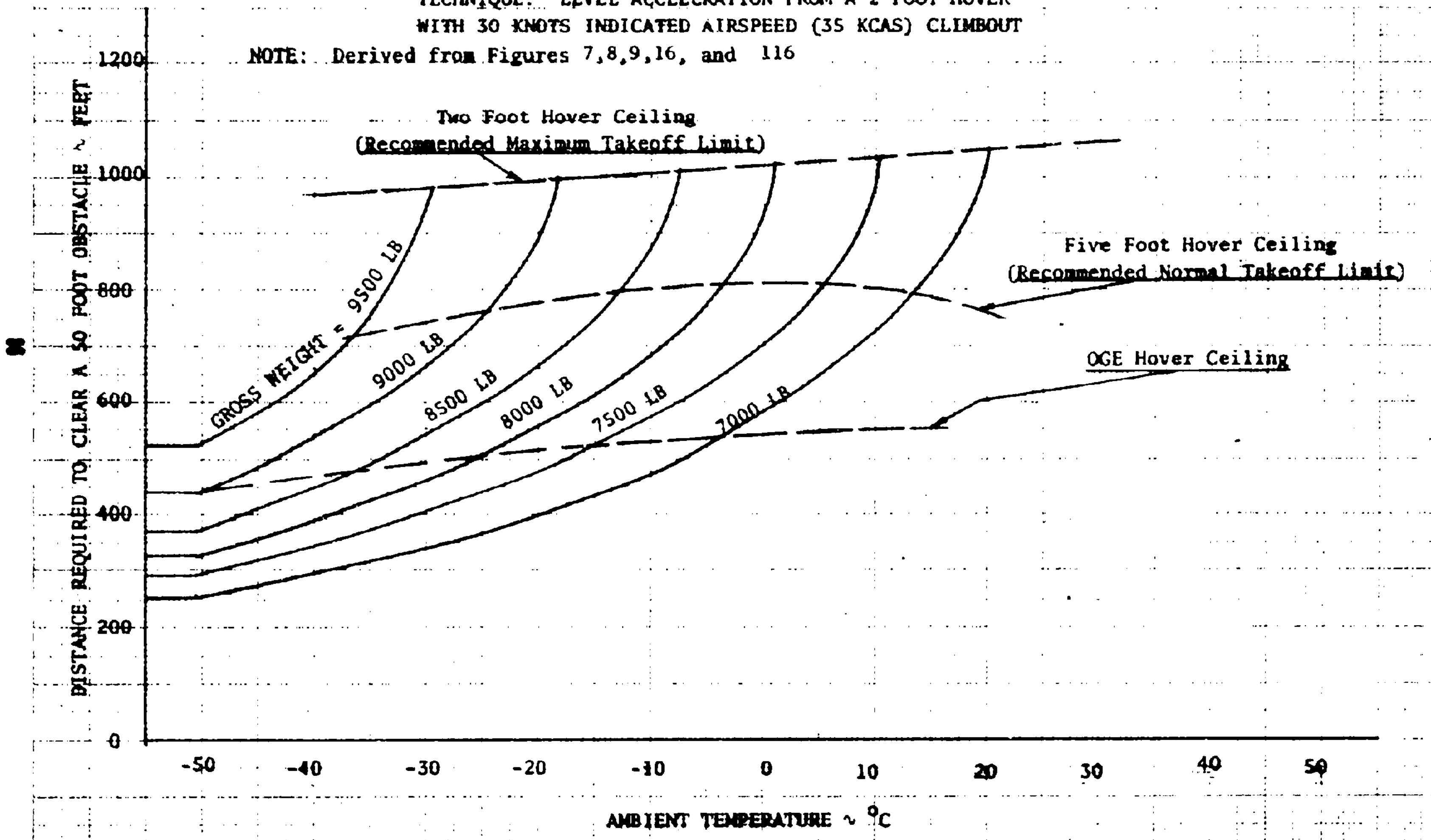


FIGURE 16

VARIATION OF TAKEOFF DISTANCE WITH CLIMBOUT AIRSPEED AND ΔC_p

YUH-1H S/N 60-6029

T53-L-13 S/N LE-14008

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER

NOTES: 1. Derived from figures 17 through 22.

2. ΔC_p = Maximum C_p available at test conditions minus C_p required to hover at 2 feet at test conditions.

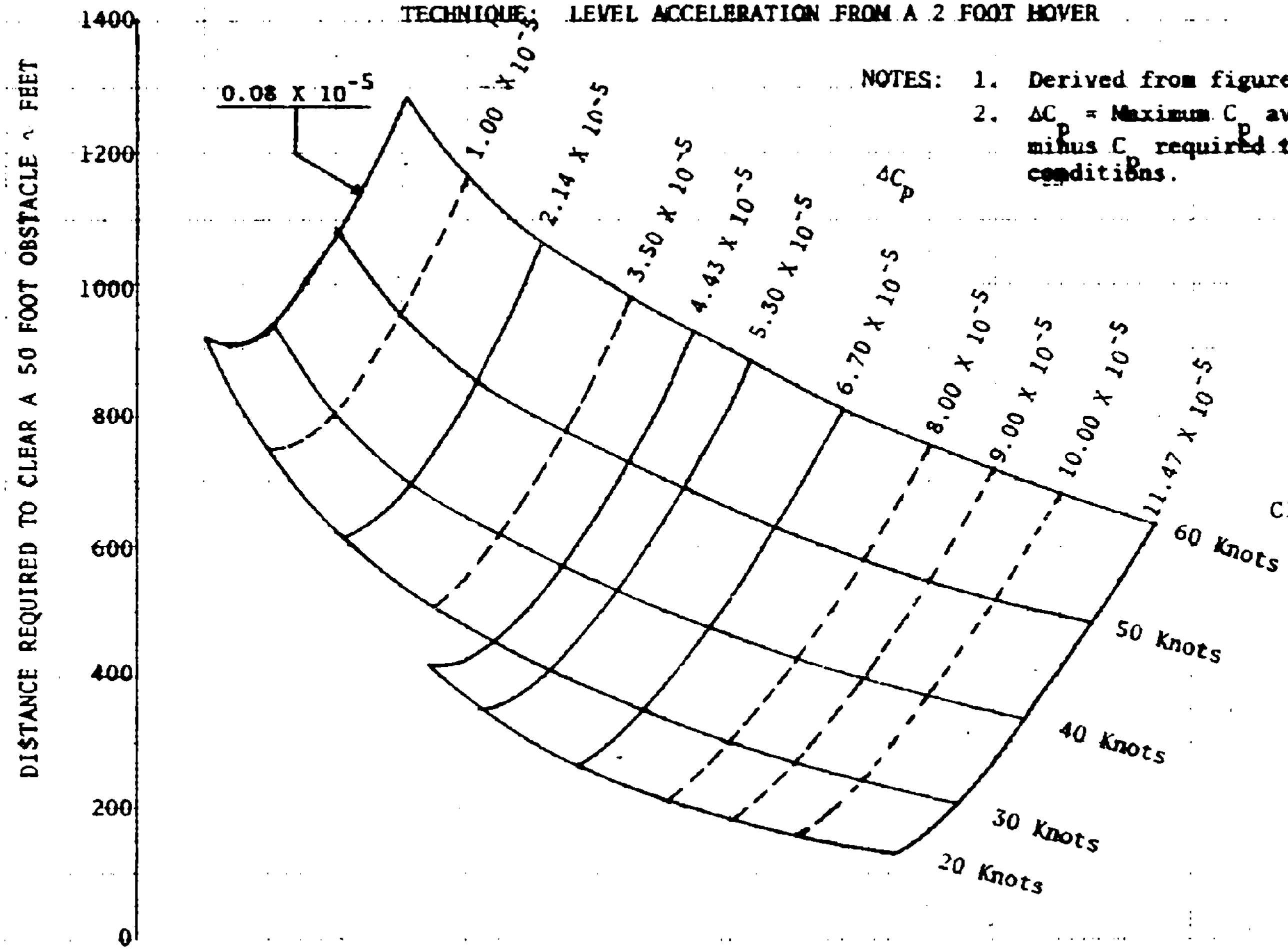


FIGURE 17
TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008
TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
MILITARY POWER
ROTOR SPEED = 324 RPM
WIND LESS THAN 3 KNOTS
AVERAGE GROSS WEIGHT = 8930 LB
AVERAGE DENSITY ALTITUDE = 13,080 FT
AVERAGE AMBIENT TEMP = 10°C
$\Delta C_p = 0.08 \times 10^{-5}$

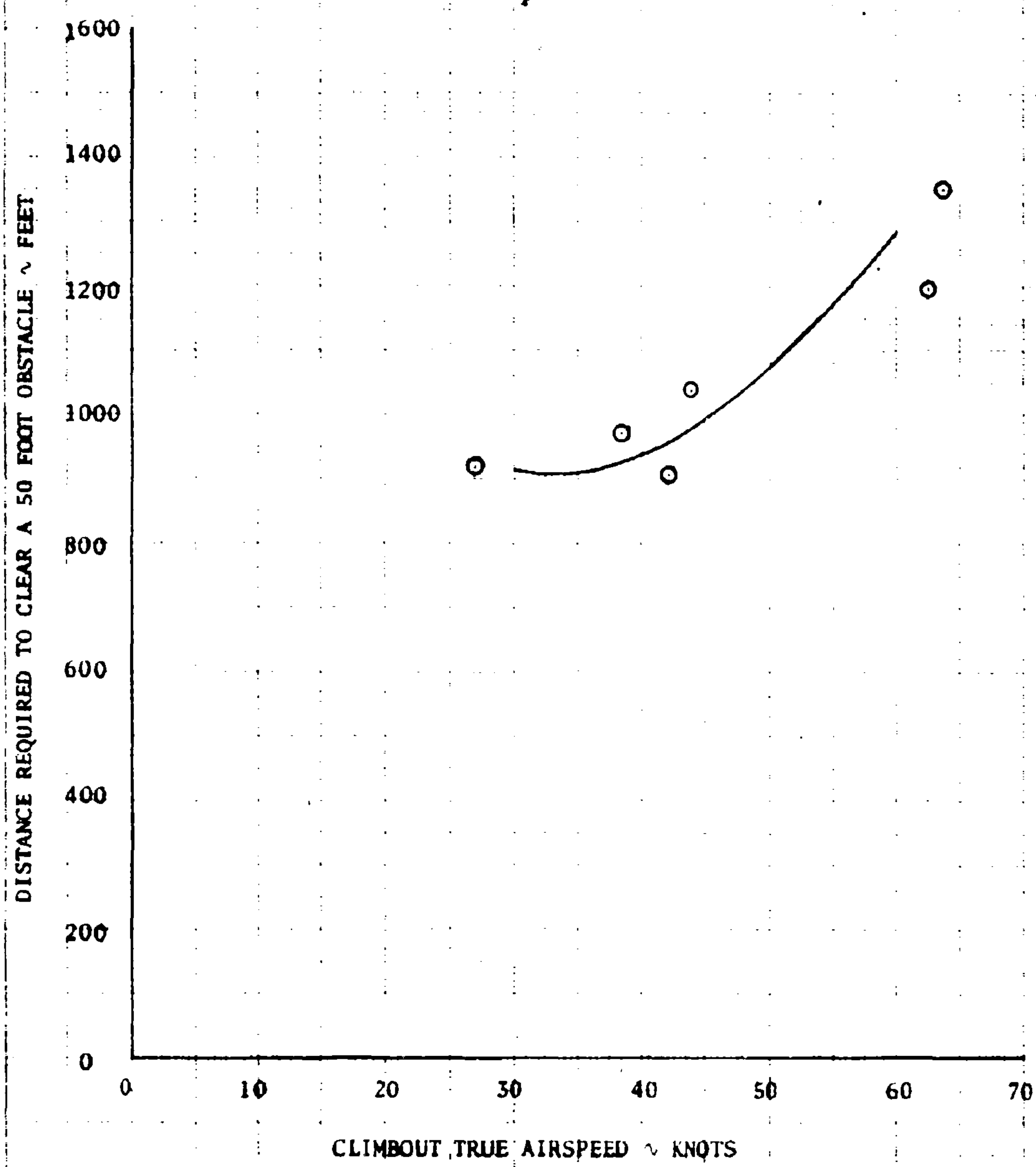


FIGURE 18

TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
MILITARY POWER
ROTOR SPEED = 324 RPM
WIND LESS THAN 3 KNOTS

AVERAGE GROSS WEIGHT = 8264 LB
AVERAGE DENSITY ALTITUDE = 13,650 FT
AVERAGE AMBIENT TEMP = 15°C
$\Delta C_p = 2.14 \times 10^{-5}$

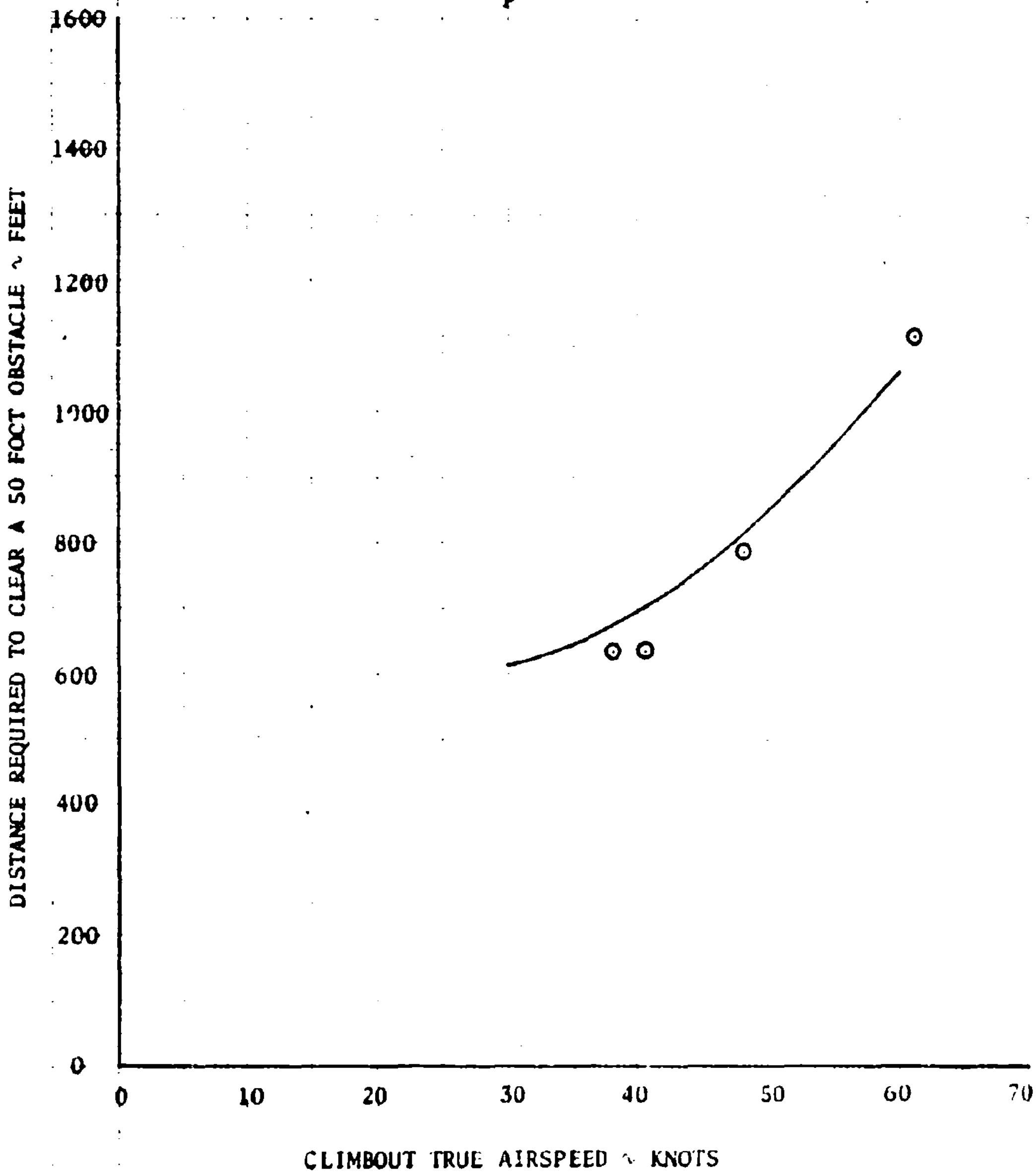


FIGURE 19
TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008
TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
MILITARY POWER
ROTOR SPEED = 324 RPM
WIND LESS THAN 3 KNOTS
AVERAGE GROSS WEIGHT = 7,880 LB
AVERAGE DENSITY ALTITUDE = 14,030 FT
AVERAGE AMBIENT TEMP = 19°C
$\Delta C_p = 4.43 \times 10^{-5}$

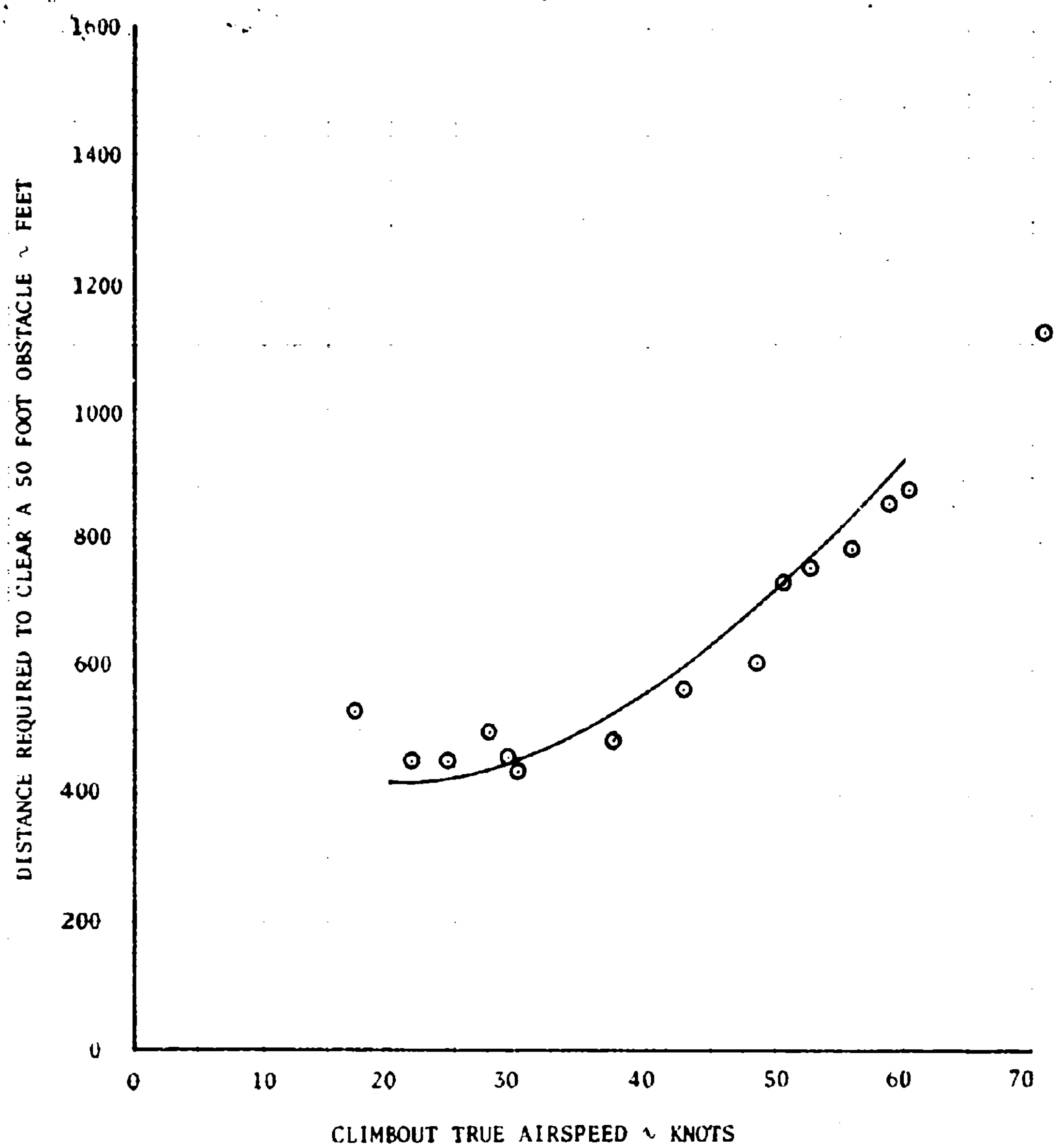


FIGURE 20
TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008
TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
MILITARY POWER
ROTOR SPEED = 324 RPM
WIND LESS THAN 3 KNOTS
AVERAGE GROSS WEIGHT = 8,640 LB
AVERAGE DENSITY ALTITUDE = 12,200 FT
AVERAGE AMBIENT TEMP = 2°C
ΔC_p = 5.30 $\times 10^{-5}$

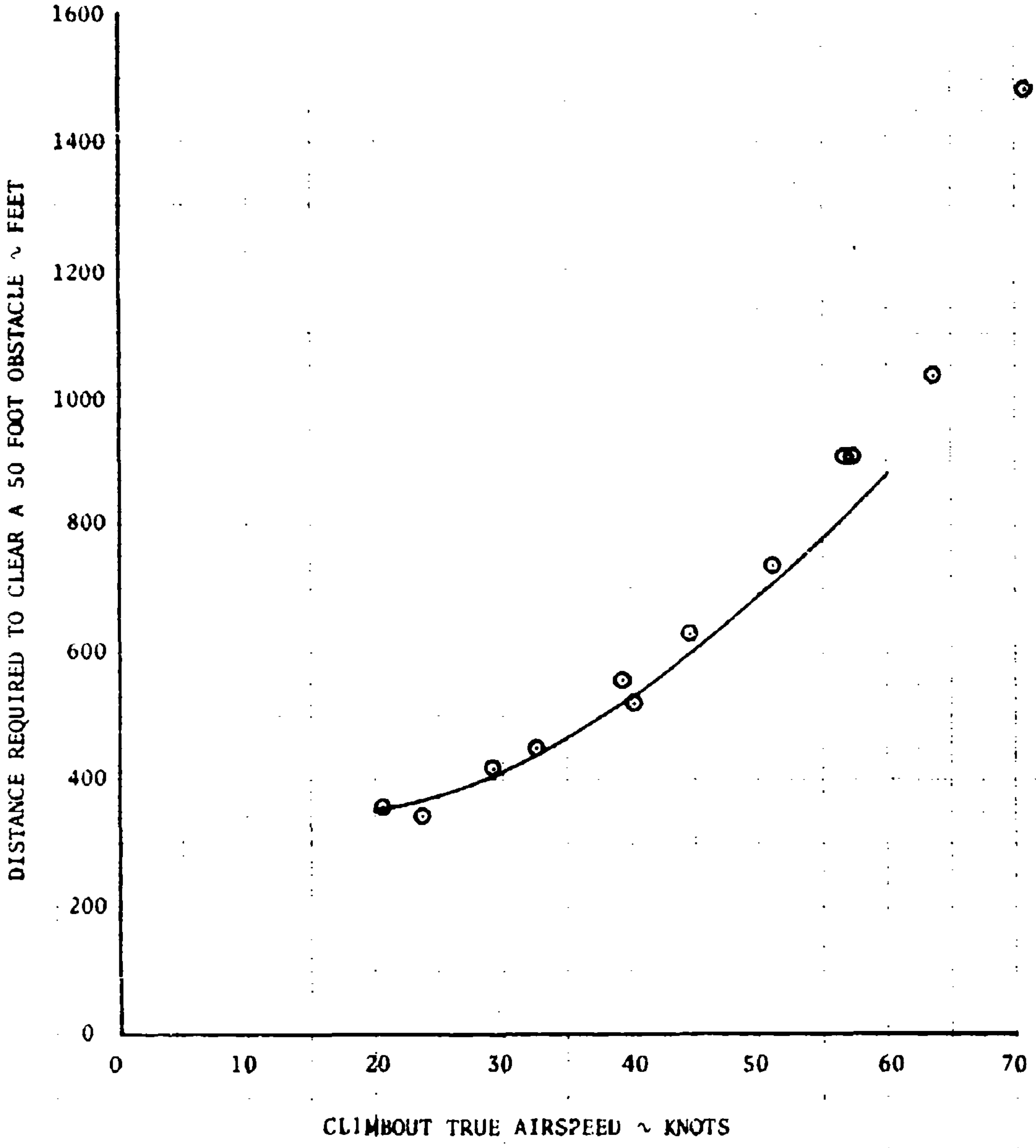


FIGURE 21
TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008

TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
MILITARY POWER
ROTOR SPEED = 324 RPM
WIND LESS THAN 3 KNOTS

AVERAGE GROSS WEIGHT = 8,750 LB
AVERAGE DENSITY ALTITUDE = 5,900 FT
AVERAGE AMBIENT TEMP = 25°C
$\Delta C_p = 6.70 \times 10^{-5}$

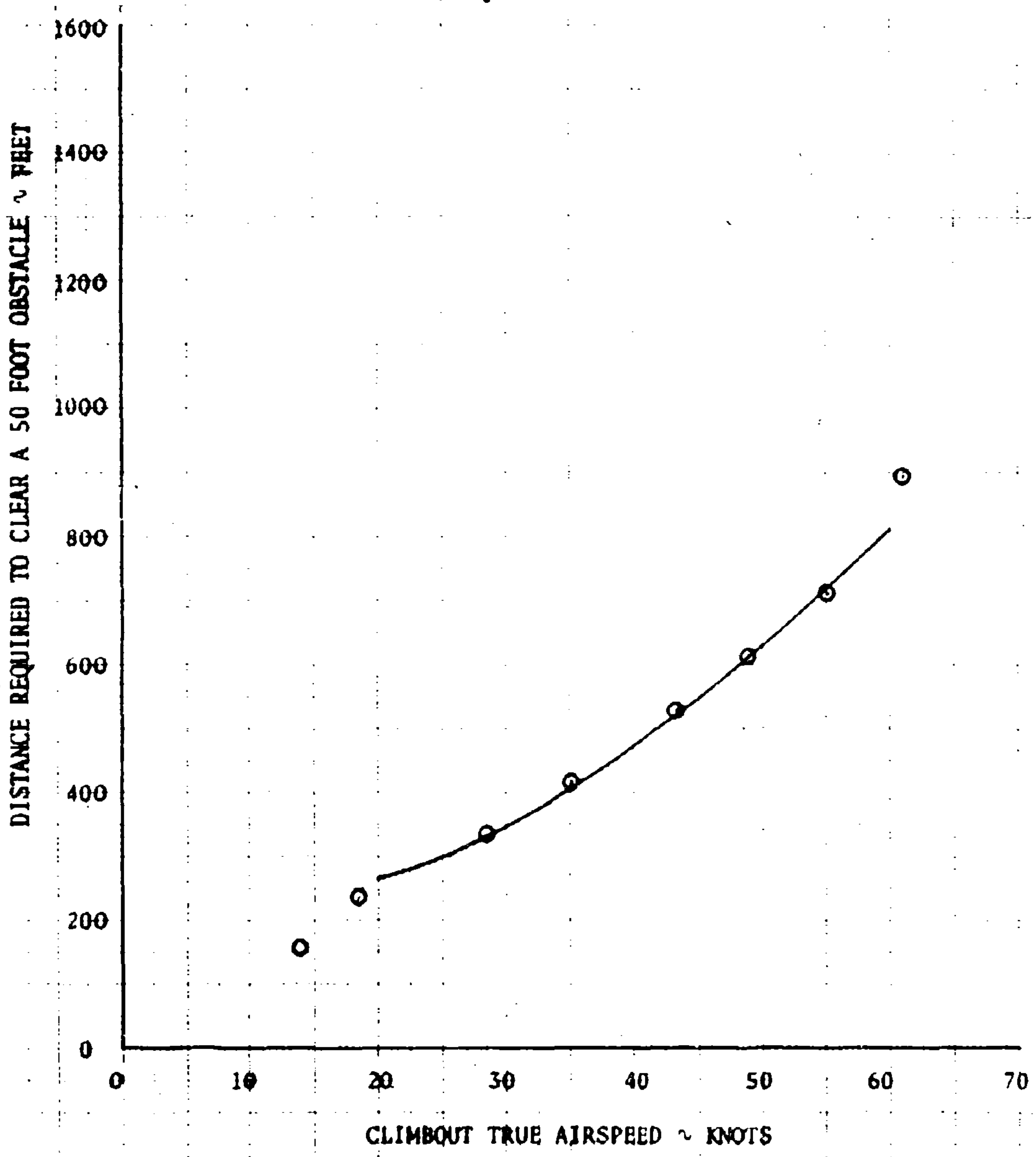


FIGURE 22
TAKEOFF DISTANCE REQUIRED
TO CLEAR A 50 FOOT OBSTACLE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008
TECHNIQUE: LEVEL ACCELERATION FROM A 2 FOOT HOVER
MILITARY POWER
ROTOR SPEED = 324 RPM
WIND LESS THAN 3 KNOTS
AVERAGE GROSS WEIGHT = 7,390 LB
AVERAGE DENSITY ALTITUDE = 6,310 FT
AVERAGE AMBIENT TEMP = 27°C
$\Delta C_p = 11.47 \times 10^{-5}$

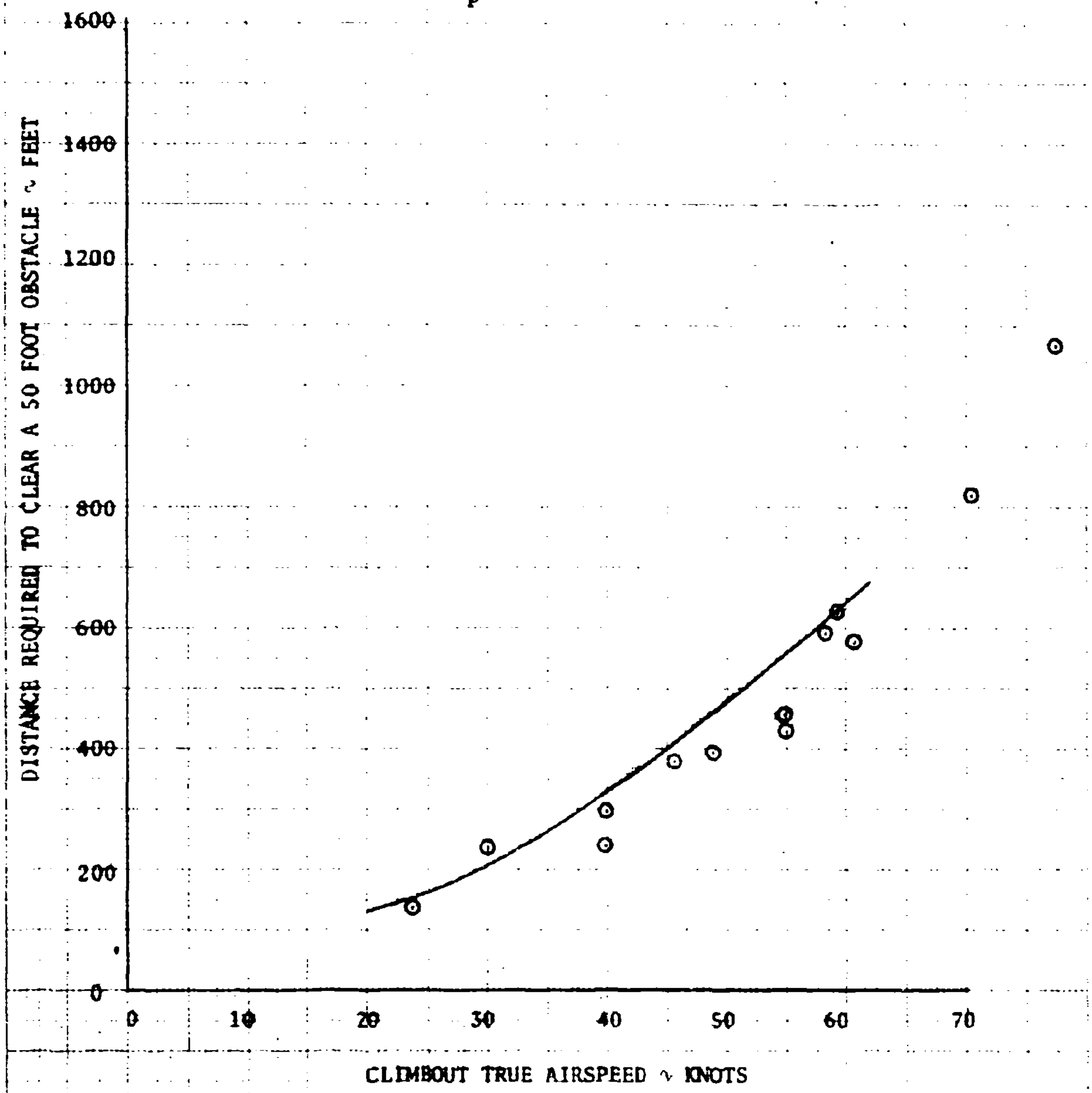


FIGURE 23
LOW AIRSPEED DECELERATION PERFORMANCE
IN GROUND EFFECT
YUH-1H S/N 60-6020

WIND LESS THAN 3 KNOTS
AVERAGE LONG. CG = 137.2 (MID)
PRESSURE ALTITUDE = 2200 FEET
AMBIENT TEMPERATURE = 12°C

SYMBOL	AVG WEIGHT	ENTRY SKID HEIGHT
○ ———	9300 LB	2 FEET
□ — —	5800 LB	2 FEET
■ — · —	5800 LB	50 FEET

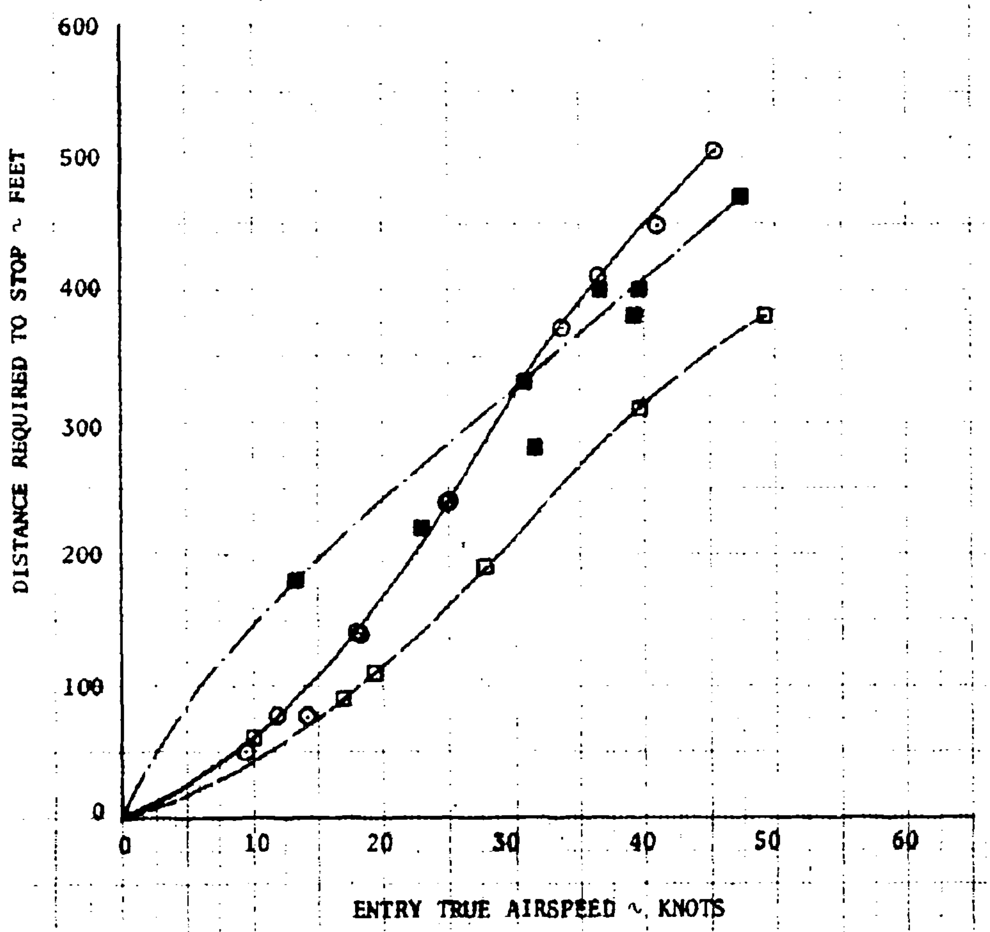


FIGURE 24

LOW AIRSPEED DECELERATION HISTORY

YUH-1H S/N 60-6029

WIND LESS THAN 3 KNOTS
PRESSURE ALTITUDE = 2200 FEET
GROSS WEIGHT = 5800 LB

ENTRY SKID HEIGHT = 2 FEET
AMBIENT TEMPERATURE = 12°C
LONGITUDINAL CG = 137.2 (MID)

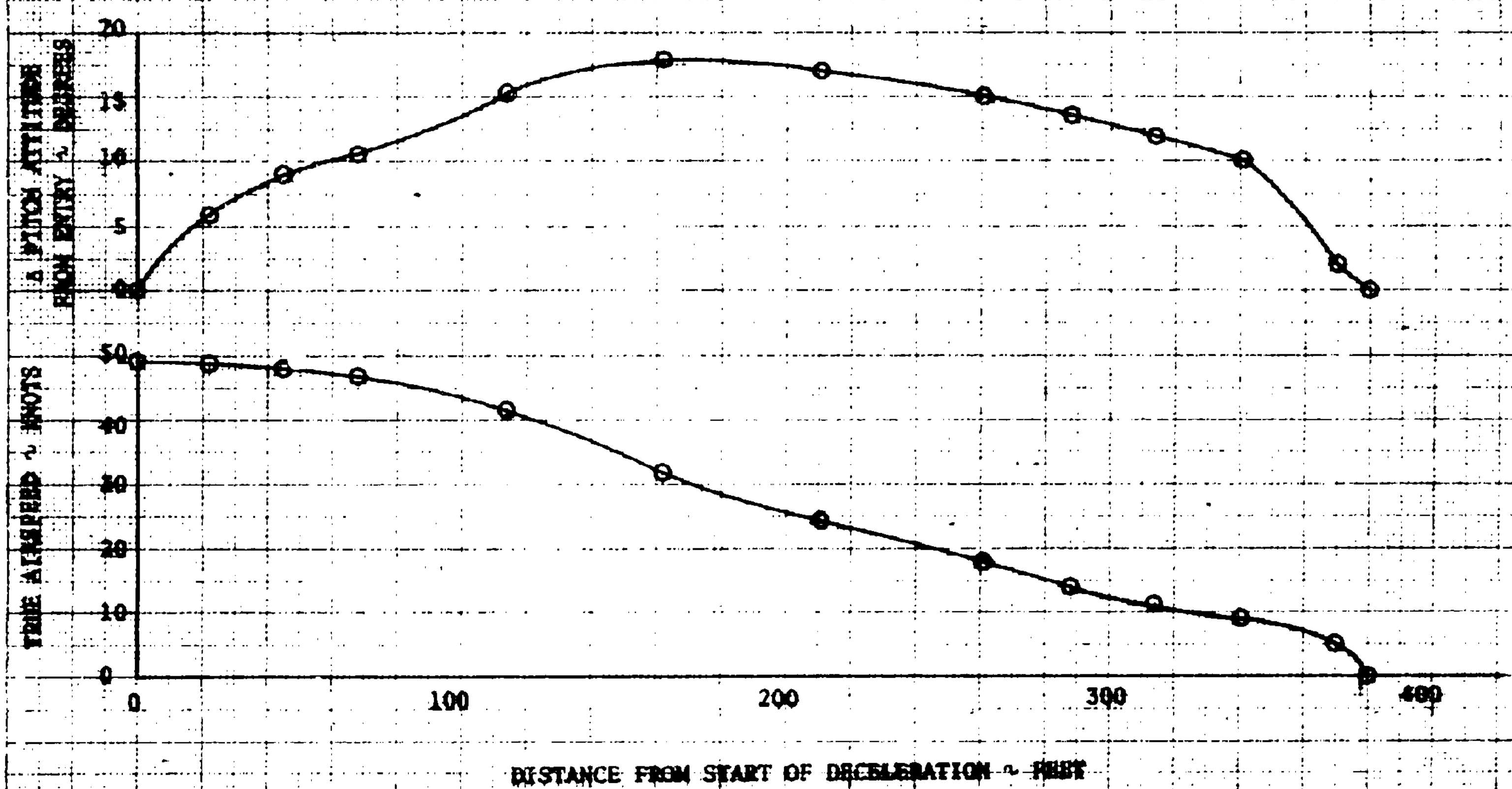


FIGURE 25
MAXIMUM SERVICE CEILING
UH-1H T53-L-13
MILITARY POWER 324 ROTOR RPM

NOTE: Derived from Figures 35,36,37,66,67 and 116.

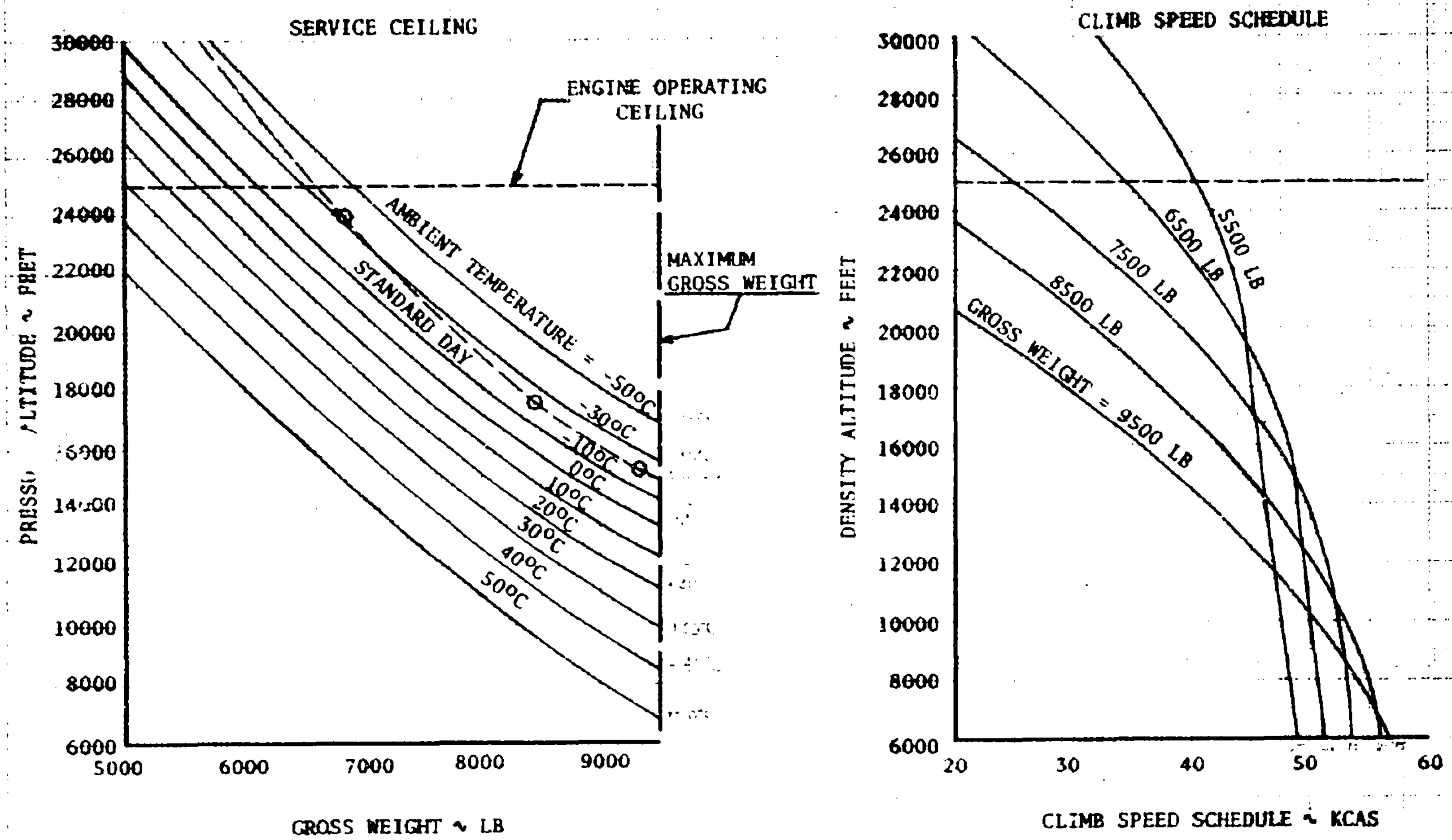


FIGURE 26

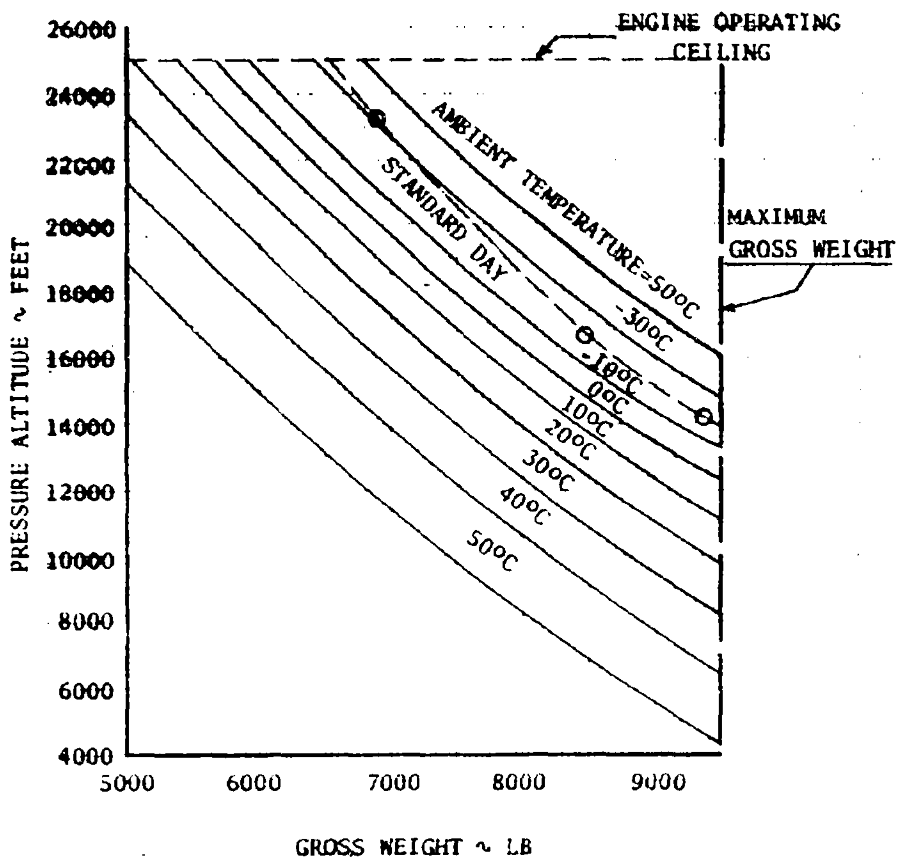
COMBAT CEILING (RATE OF CLIMB = 500 FT/MIN)

UH-1H T53-L-13

MILITARY POWER 324 ROTOR RPM

NOTE: Derived from Figures 35,36,37,66,67 and 116

COMBAT CEILING



CLIMB SPEED SCHEDULE

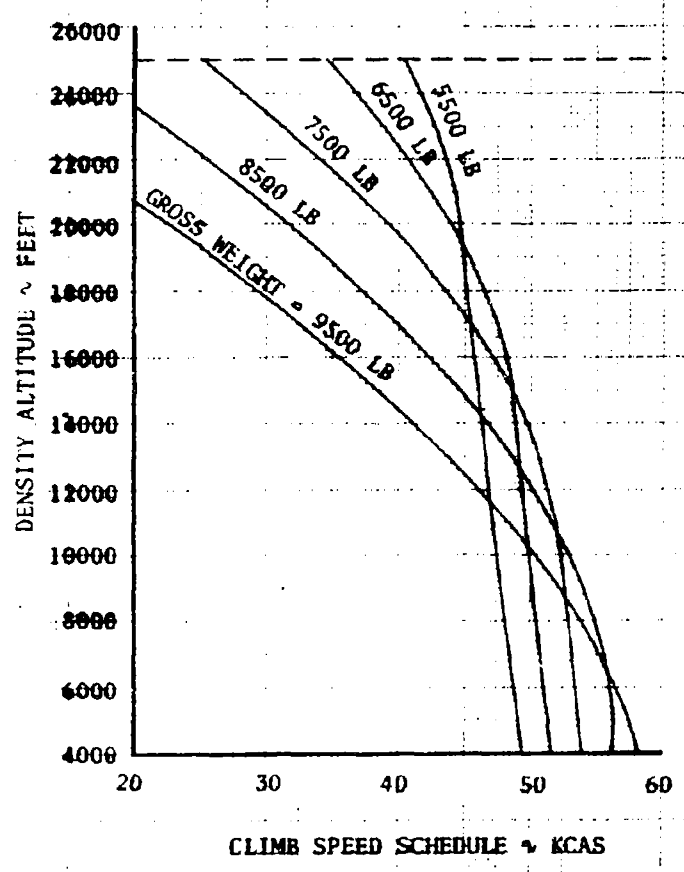


FIGURE 27
MAXIMUM CLIMB PERFORMANCE SUMMARY
UH-1H T53-L-13
STANDARD DAY MILITARY POWER 324 ROTOR RPM

NOTE: Derived from Figures 35,36,37,66,67,116 and 121.

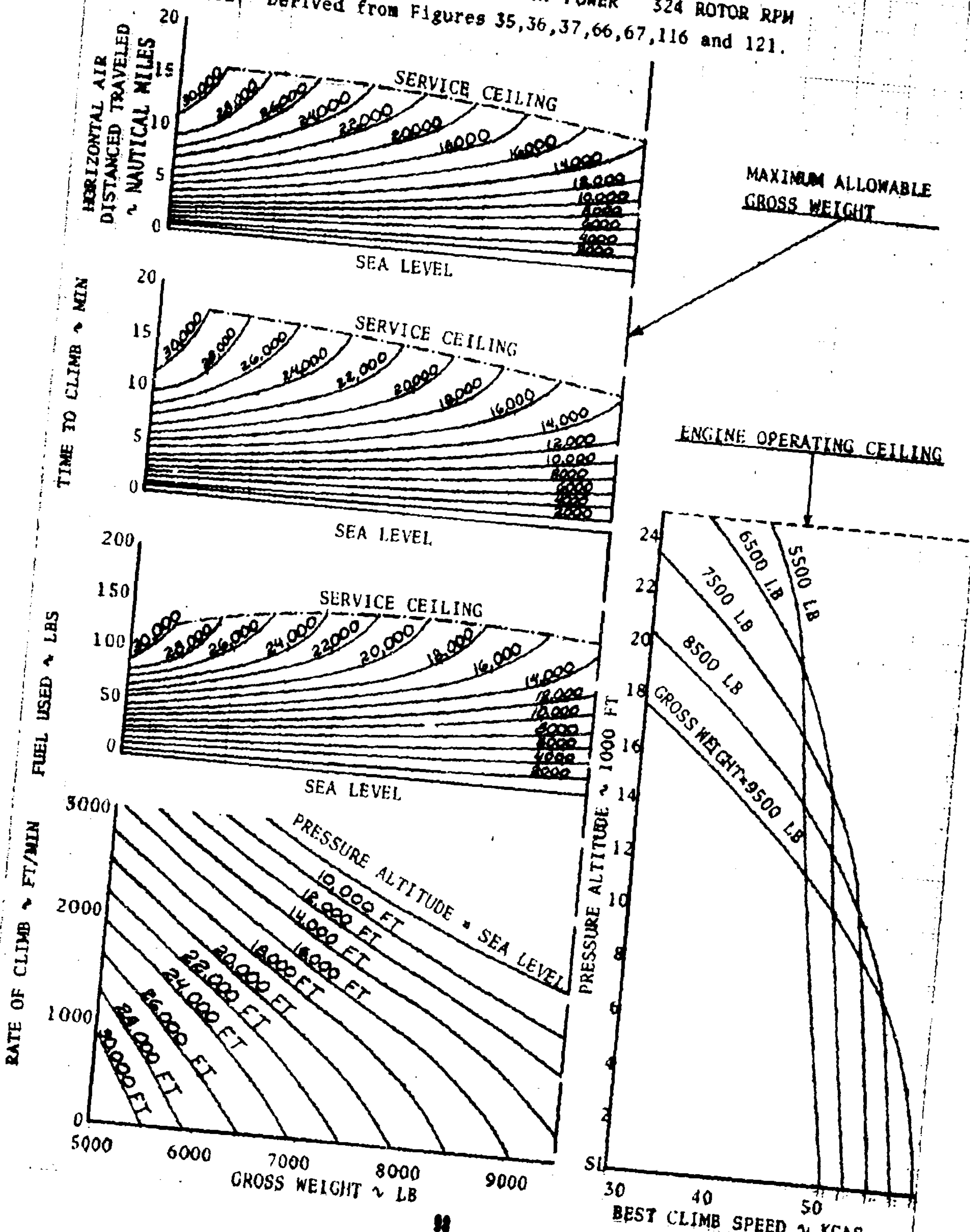
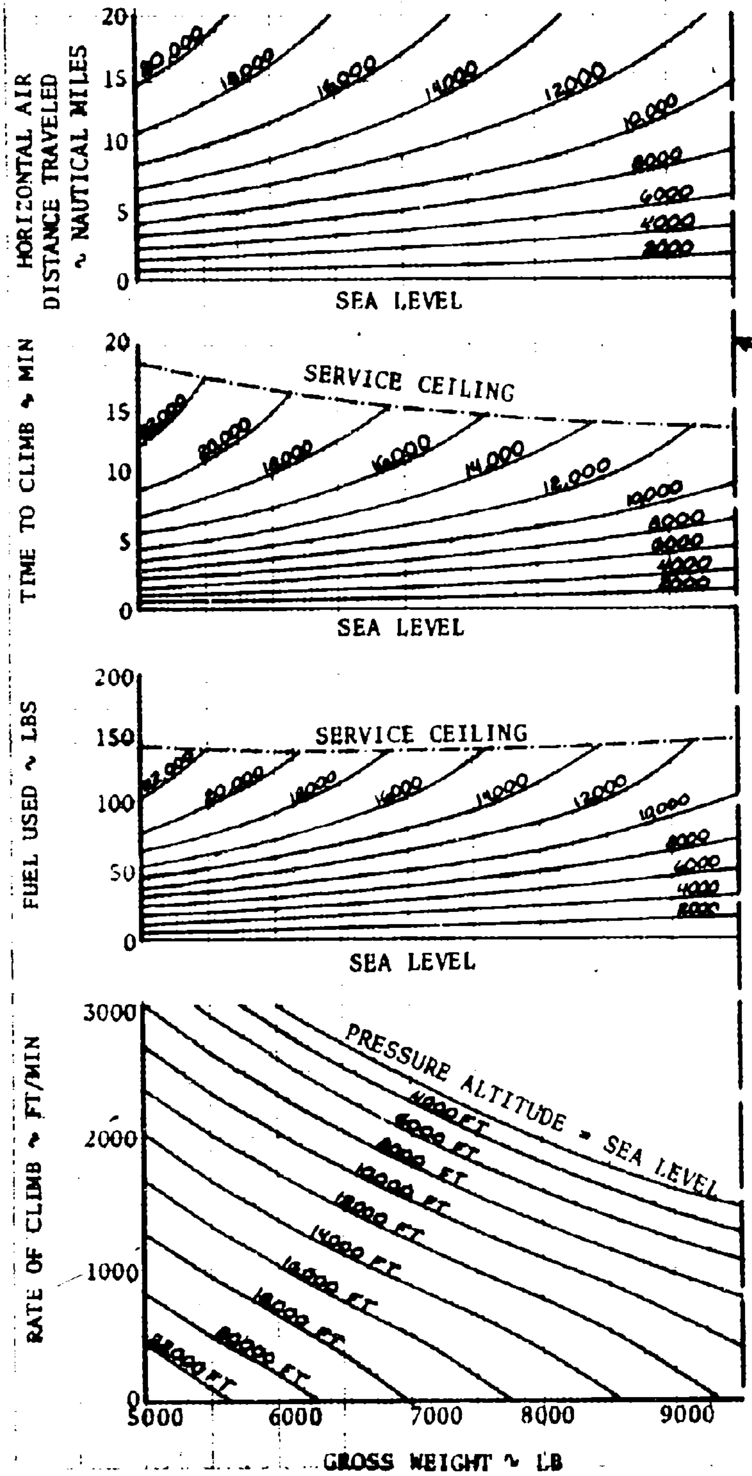


FIGURE 28
NORMAL CLIMB PERFORMANCE SUMMARY
UH-1H TS3-L-13
MAXIMUM CONTINUOUS POWER 324 ROTOR RPM
CLIMB SPEED= 80 KCAS STANDARD DAY



NOTE:

Derived From Figures 36, 37,66,67,117, and 121.

MAXIMUM ALLOWABLE GROSS WEIGHT

STANDARD DAY	
PRESSURE ALTITUDE	STANDARD TEMP ~°C
SEA LEVEL	15°C
2000	11
4000	7
6000	3
8000	-1
10000	-5
12000	-9
14000	-13
16000	-17
18000	-21
20000	-25
22000	-29
25000	-35

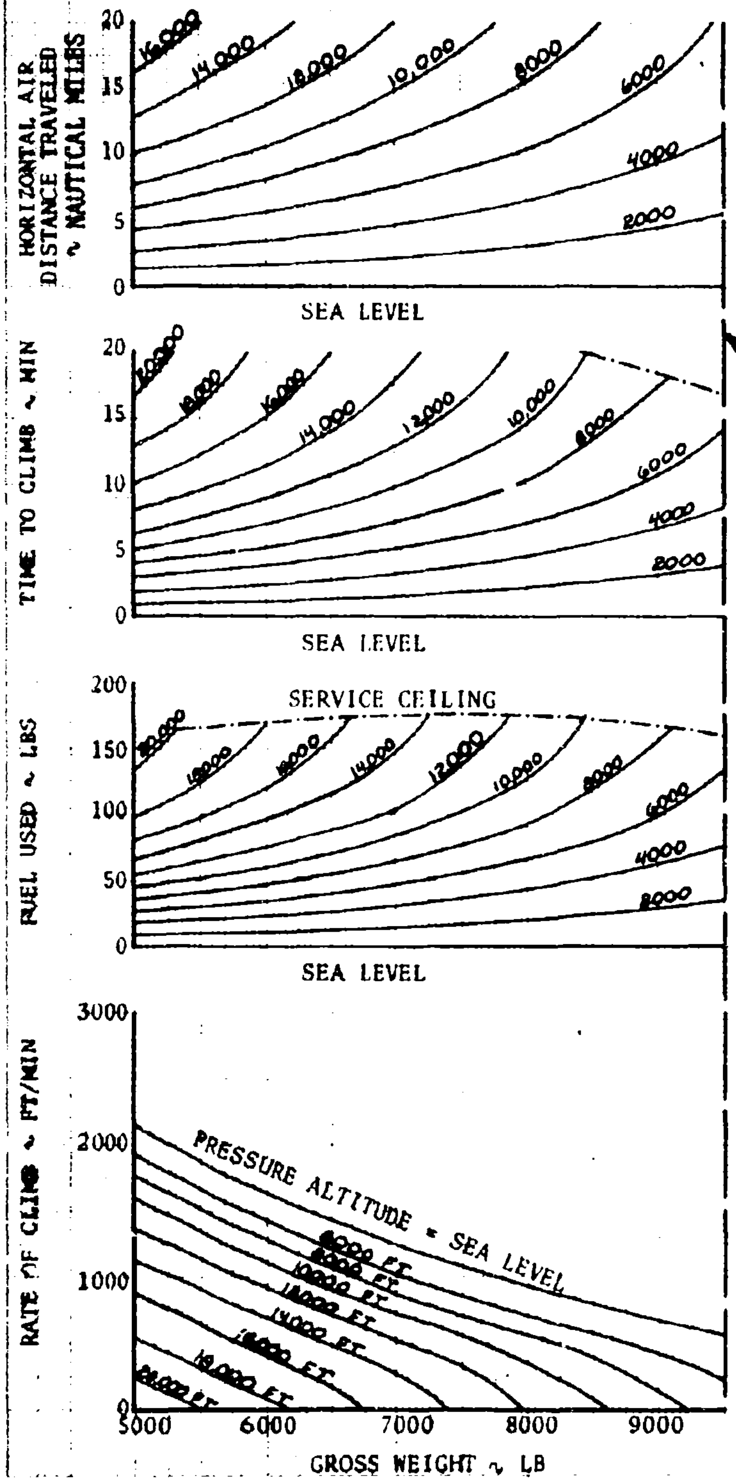
FIGURE 29

NORMAL CLIMB PERFORMANCE SUMMARY

UH-1H T53-L-13

MAXIMUM CONTINUOUS POWER 324 ROTOR RPM

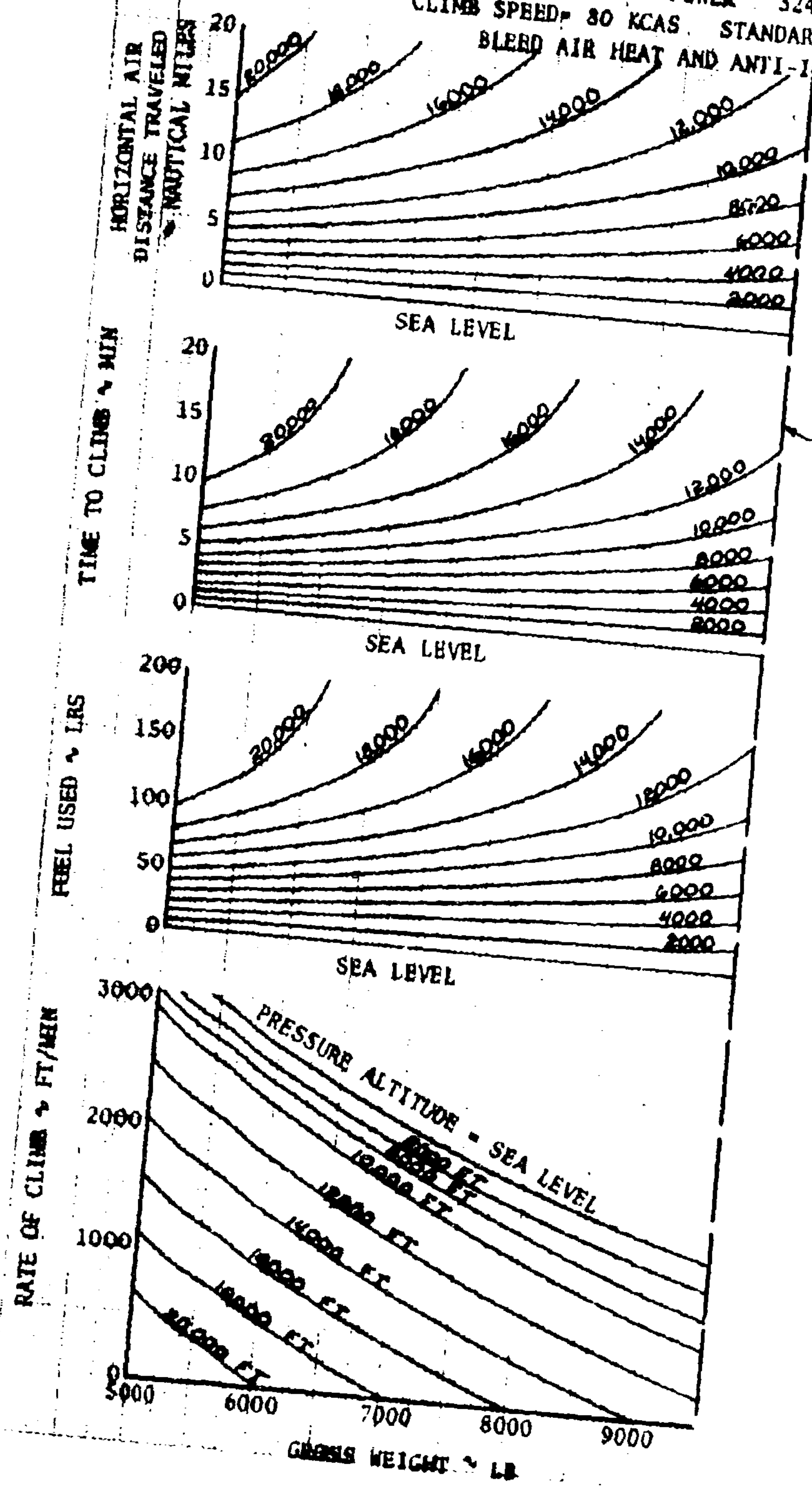
CLIMB SPEED= 80 KCAS STANDARD DAY +30°C



NOTE:
Derived From Figures 36, 37, 66, 67, and specification power and fuel flow corrected for installation losses on figure 116.

STANDARD DAY +30°C	
PRESSURE ALTITUDE	STANDARD TEMP +30°C
SEA LEVEL	45°C
2000	41
4000	37
6000	33
8000	29
10000	25
12000	21
14000	17
16000	13
18000	9
20000	5
22000	1
25000	-5

FIGURE 30
NORMAL CLIMB PERFORMANCE SUMMARY
UH-1H T53-L-13
MAXIMUM CONTINUOUS POWER 324 ROTOR RPM
CLIMB SPEED= 80 KCAS STANDARD DAY -30°C
BLEED AIR HEAT AND ANTI-ICE ON



NOTE:
Derived From Figures 36, 37,66,67, and specification power and fuel flow corrected for installation losses on figure 120

STANDARD DAY -30°C	
PRESSURE ALTITUDE	STANDARD TEMP -30°C
SEA LEVEL	-15°C
2000	-19
4000	-23
6000	-27
8000	-31
10000	-35
12000	-39
14000	-43
16000	-47
18000	-51
20000	-55
22000	-59
25000	-65

FIGURE 31
CLIMB PERFORMANCE
YUH-1H S/N 60-6029
MILITARY POWER 324 ROTOR RPM

STANDARD DAY

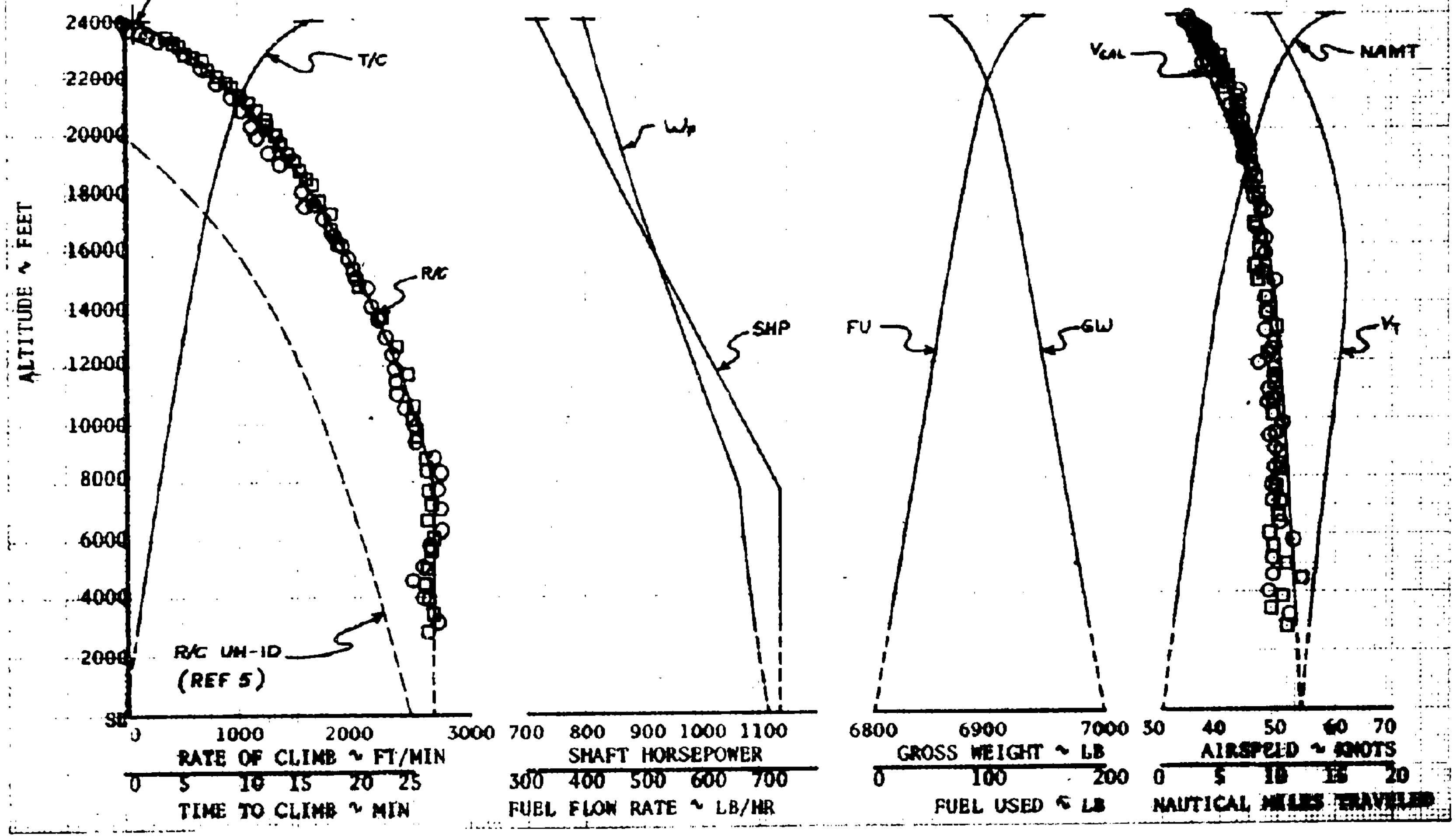


FIGURE 32
CLIMB PERFORMANCE
YUH-1H S/N 60-6029
MILITARY POWER 324 ROTOR RPM

STANDARD DAY

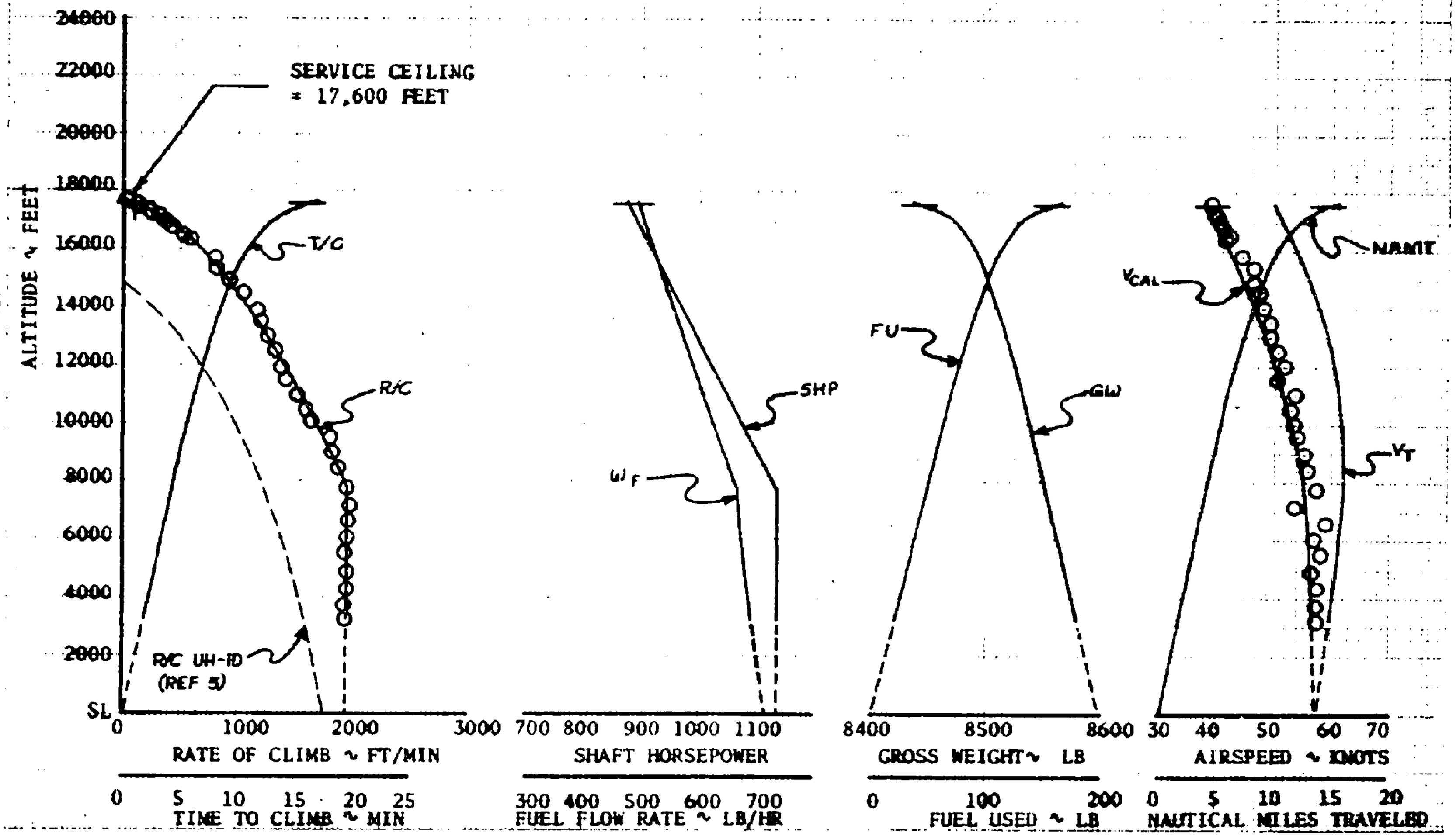


FIGURE 33
CLIMB PERFORMANCE
YUH-1H S/N 60-6029
MILITARY POWER 324 ROTOR RPM

STANDARD DAY

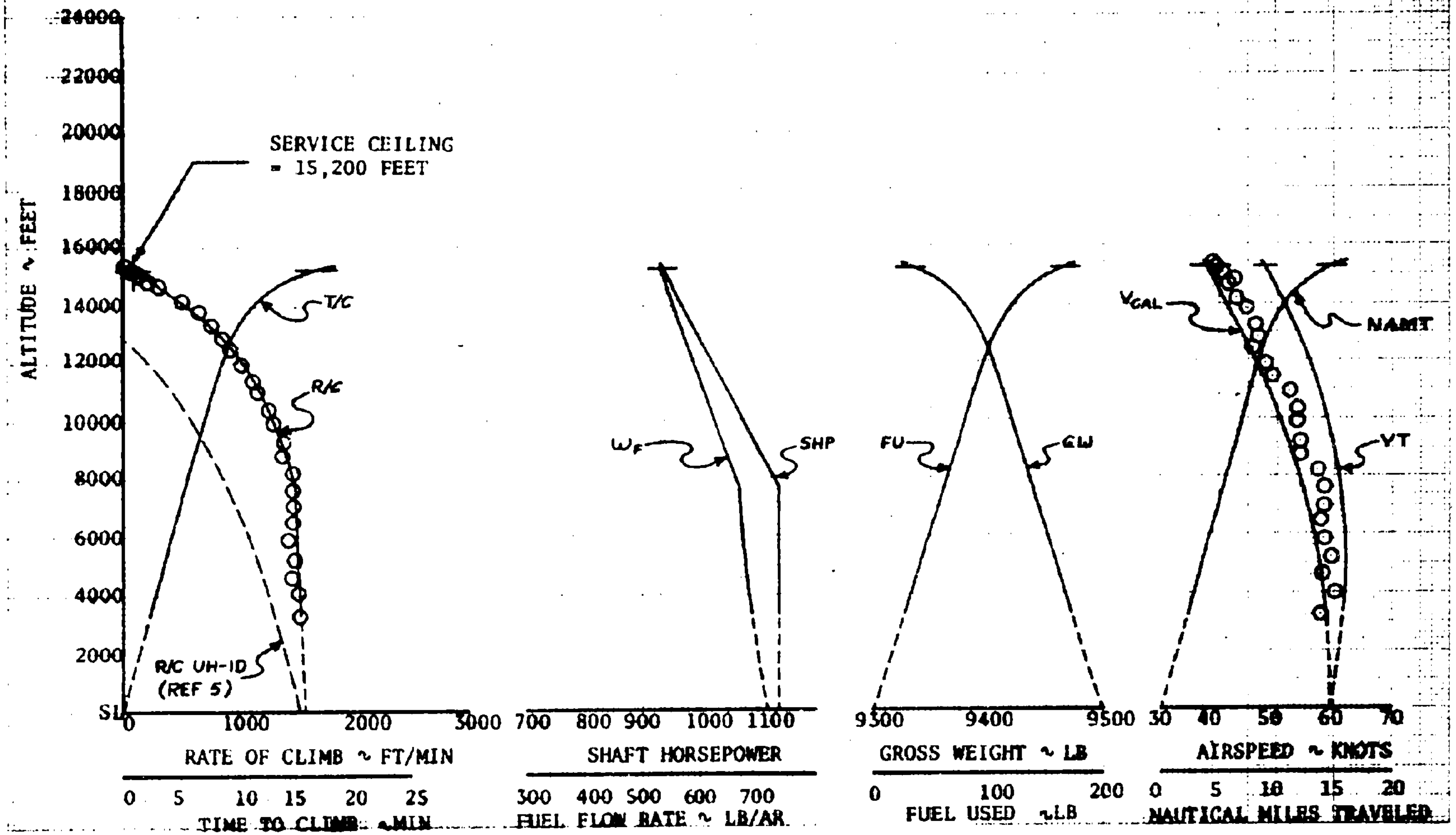


FIGURE 34
"NORMAL" CLIMB PERFORMANCE
YUH-1H S/N 60-6029
MAXIMUM CONTINUOUS POWER 324 ROTOR RPM

STANDARD DAY

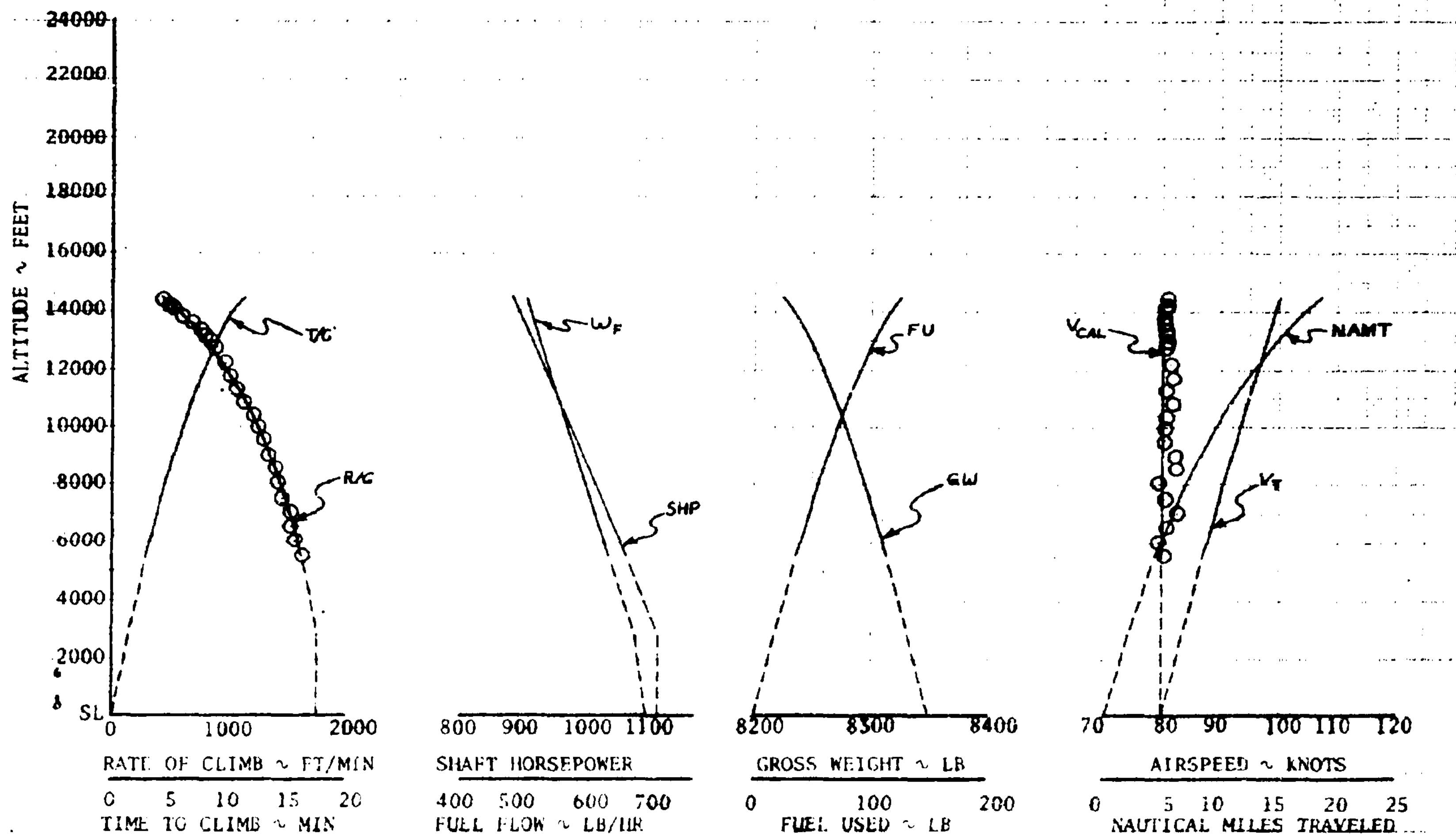
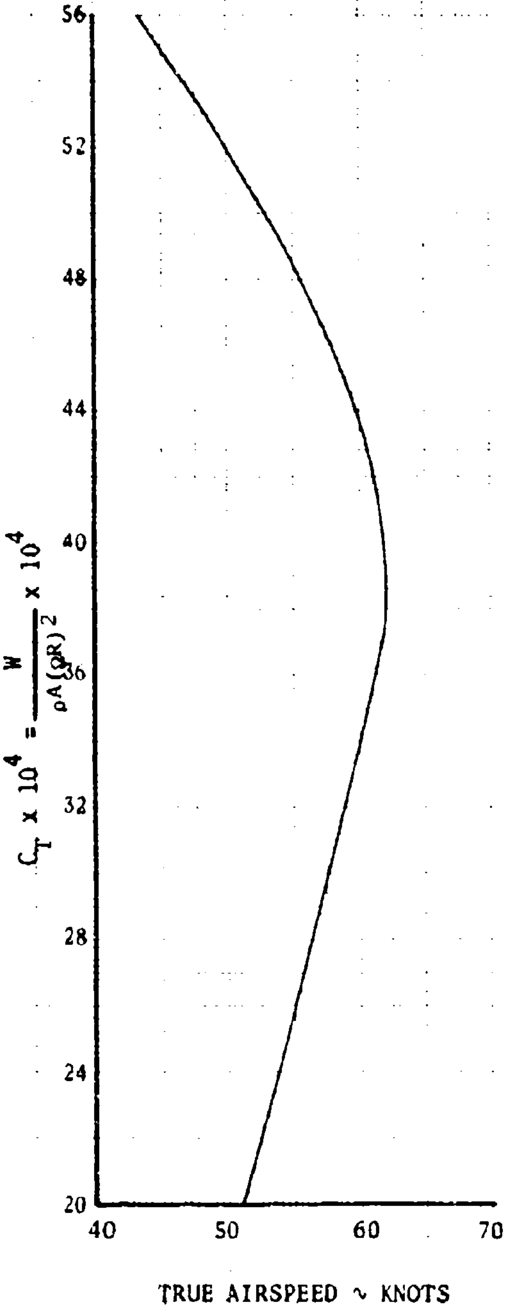


FIGURE 35
CLIMB SPEED SCHEDULE
YUH-1H S/N 60-6029

NOTE: This airspeed schedule was derived from figures 66, 67, and sawtooth climb data not in this report. An example of an airspeed variation sawtooth climb is shown below right.



AIRSPEED SAWTOOTH
SAMPLE
ROTOR SPEED = 324 RPM
GROSS WEIGHT = 9111 LB
DENSITY ALTITUDE = 15,000 FT

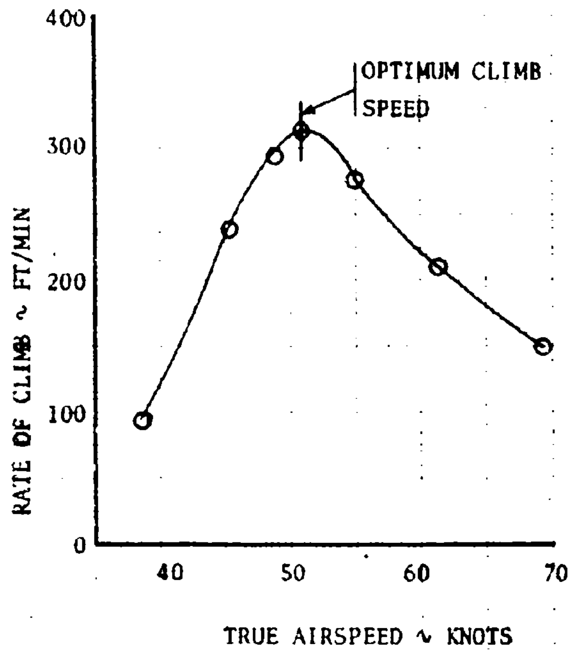


FIGURE 36
EFFECT OF GROSS WEIGHT AND ROTOR SPEED
ON CLIMB PERFORMANCE
YUH-1H S/N 60-6029

NOTE: Tests conducted at figure 35 climb speed.

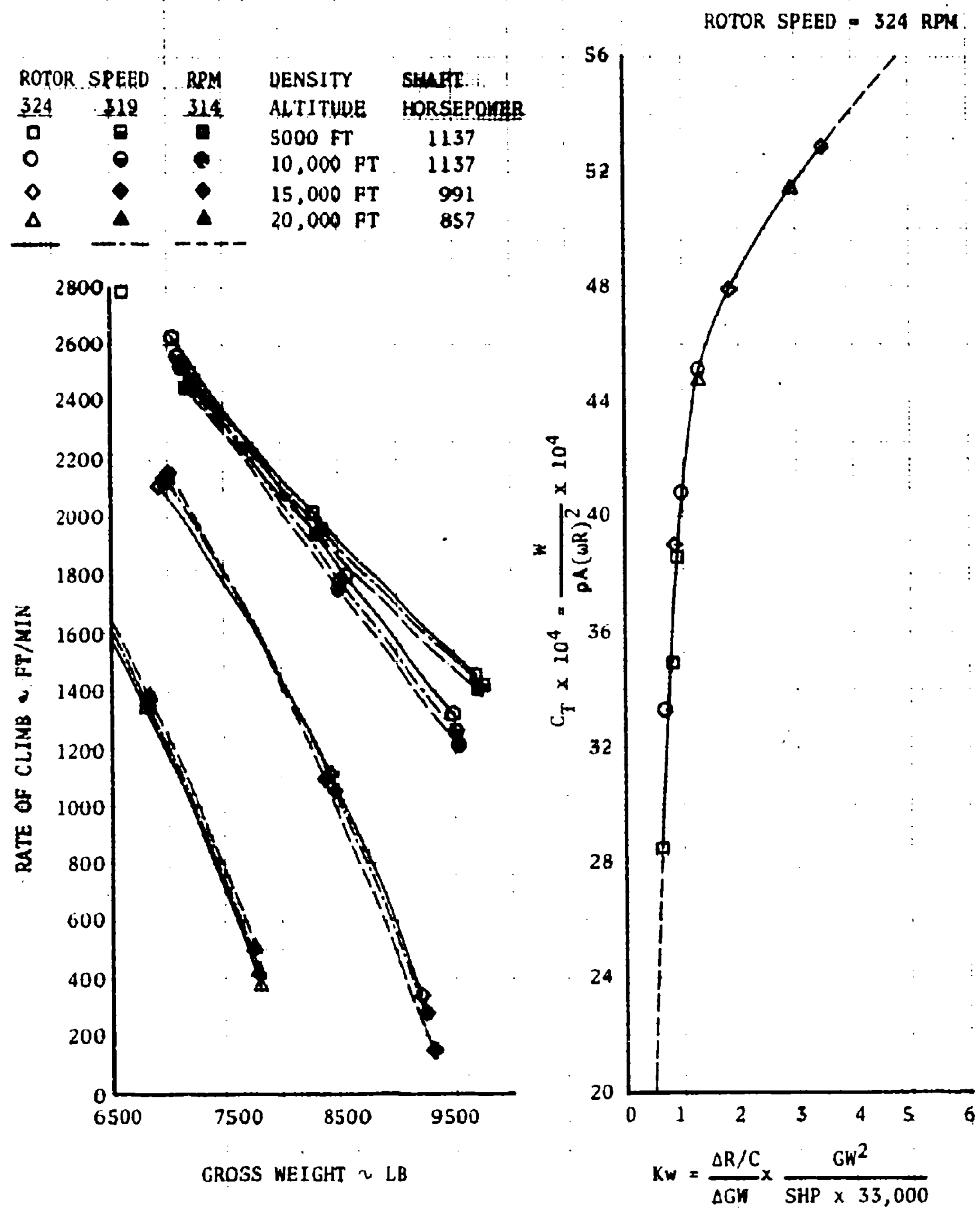
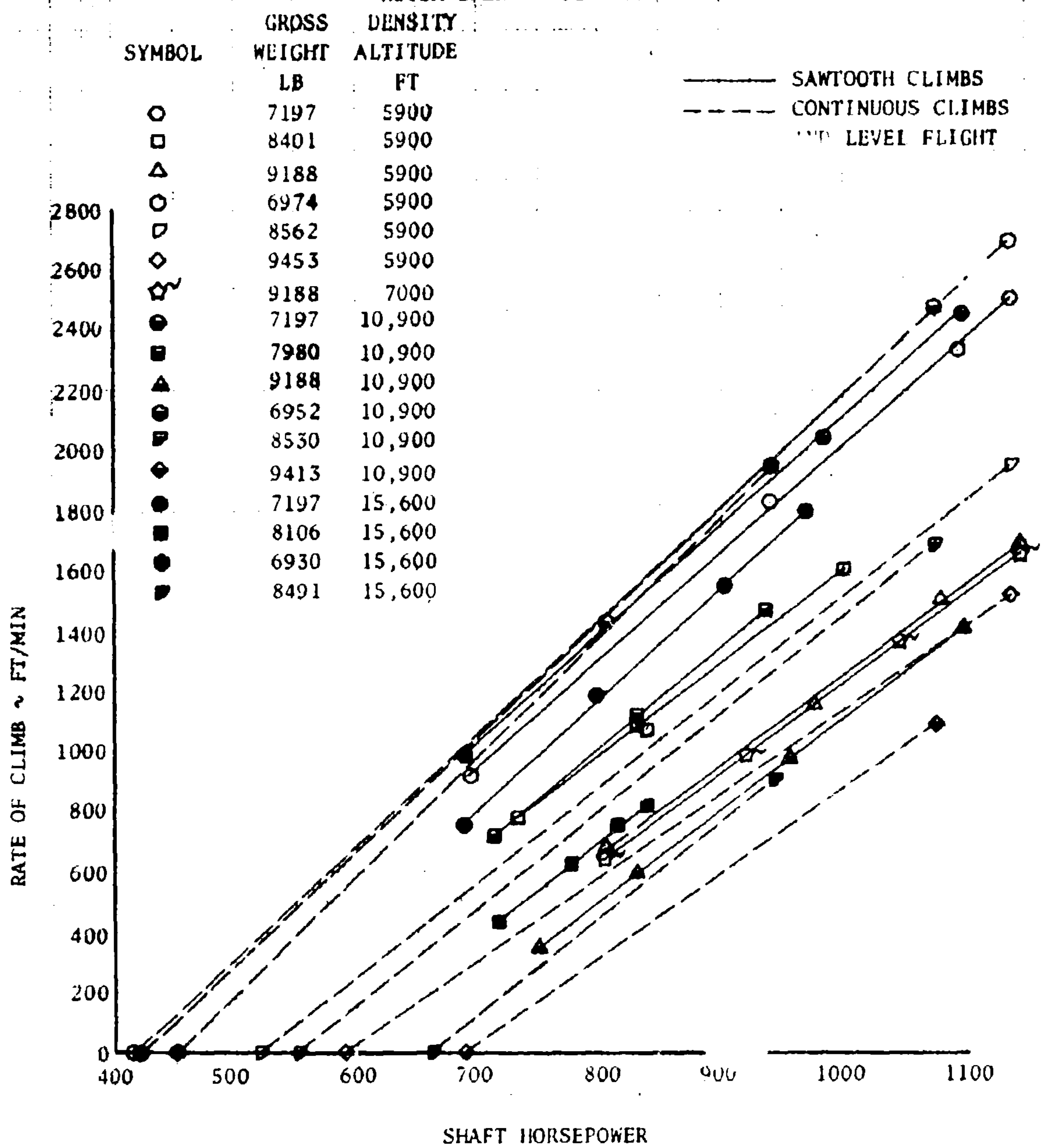


FIGURE 37
EFFECT OF POWER VARIATION ON CLIMB PERFORMANCE
YUH-1H S/N 60-6029

NOTE: Tests conducted at figure 35 climb speed.

ROTOR SPEED = 324 RPM



$$Kp = \frac{\Delta R/C}{\Delta SHP} \times \frac{GW}{33,000}$$

$.77 < Kp < .84$ NO TREND WITH WEIGHT OR ALTITUDE
USE $Kp = .80$

FIGURE 38

OPTIMUM ALTITUDE CRUISE

UH-1H T53-L-13

STANDARD DAY CRUISE SPEED=V_{NE}

NOTE: Derived from Figures 39 and 40

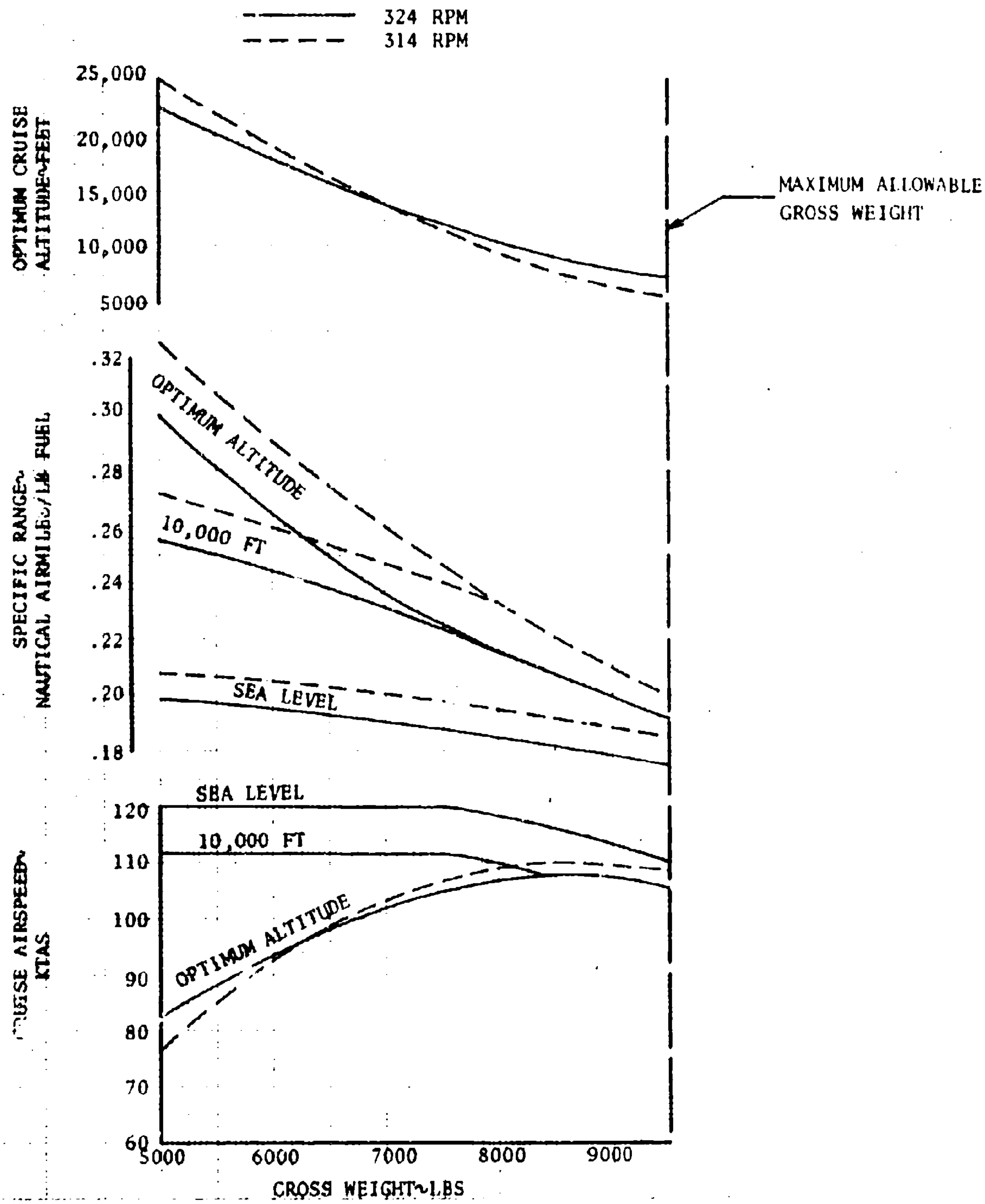


FIGURE 39
RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM STANDARD DAY CLEAN CONFIGURATION

NOTE: Derived from Figures 66,67,117,121 and 131

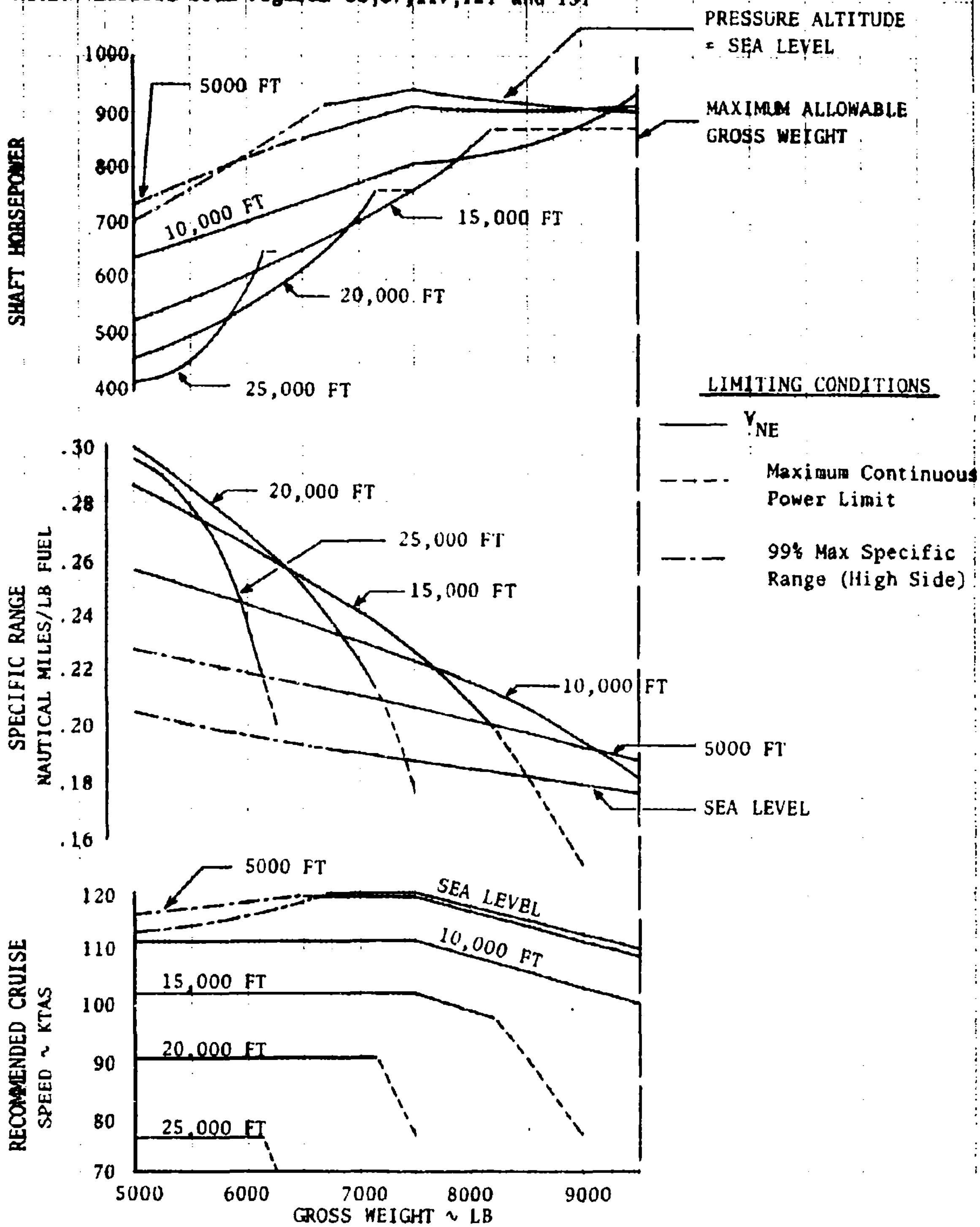


FIGURE 39 Continued
RANGE PERFORMANCE
UH-1H TS3-L-13
324 ROTOR RPM STANDARD DAY

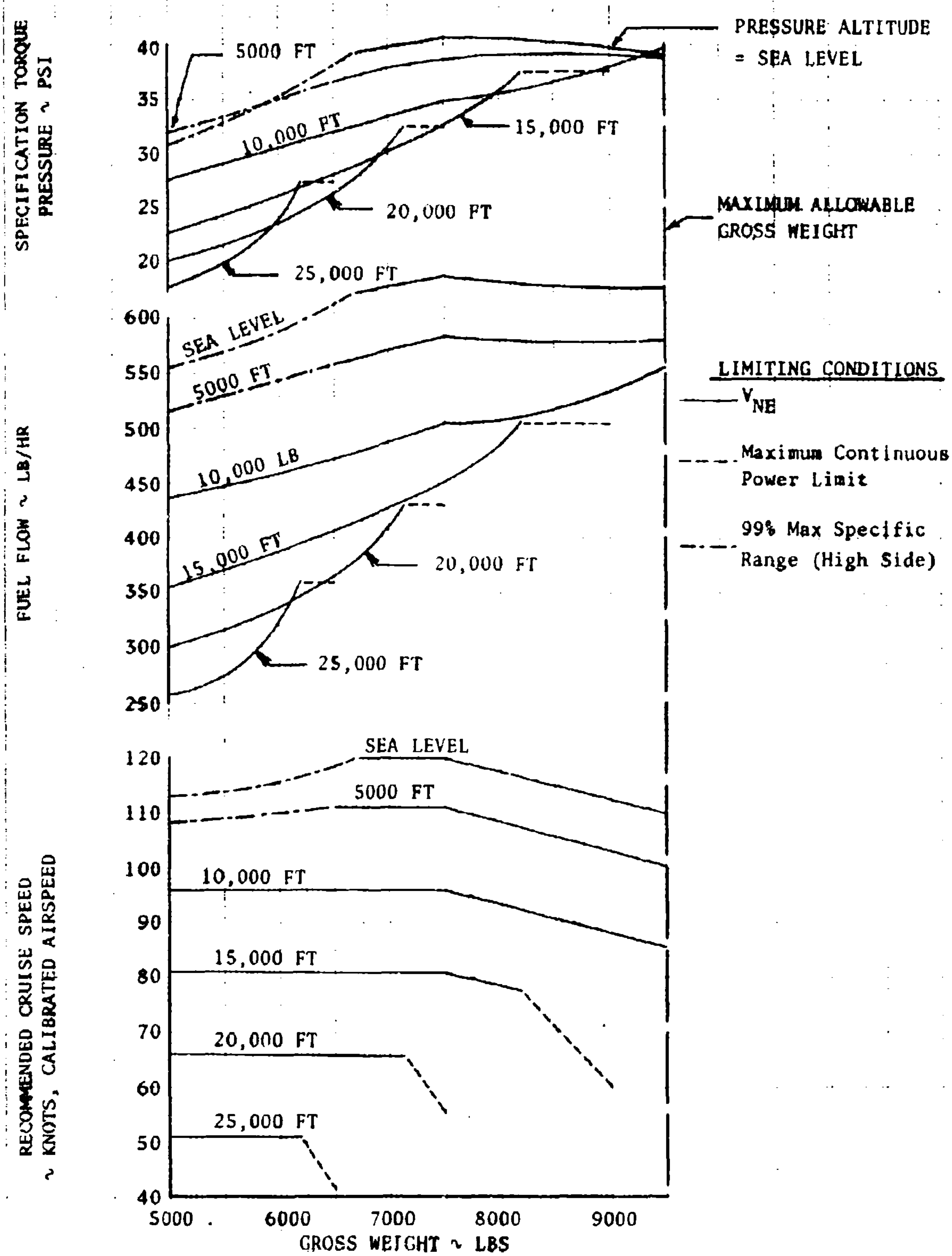


FIGURE 40
RANGE PERFORMANCE
UH-1H T53-L-13
314 ROTOR RPM STANDARD DAY CLEAN CONFIGURATION

NOTE: Derived from Figures 66,67,118,131 and corrected specification fuel flow

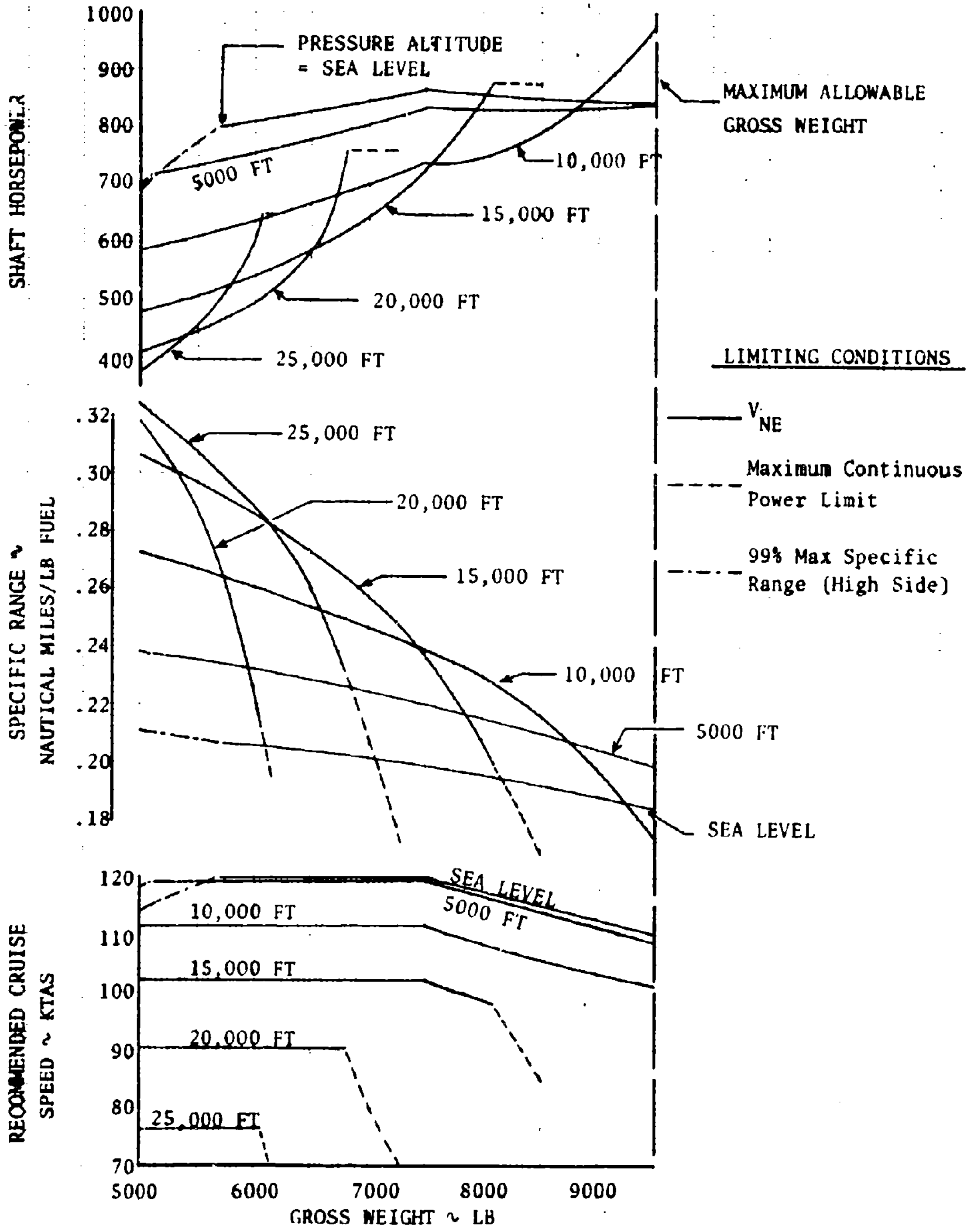


FIGURE 40 Continued
RANGE PERFORMANCE
UH-1H T53-L-13
314 ROTOR RPM STANDARD DAY

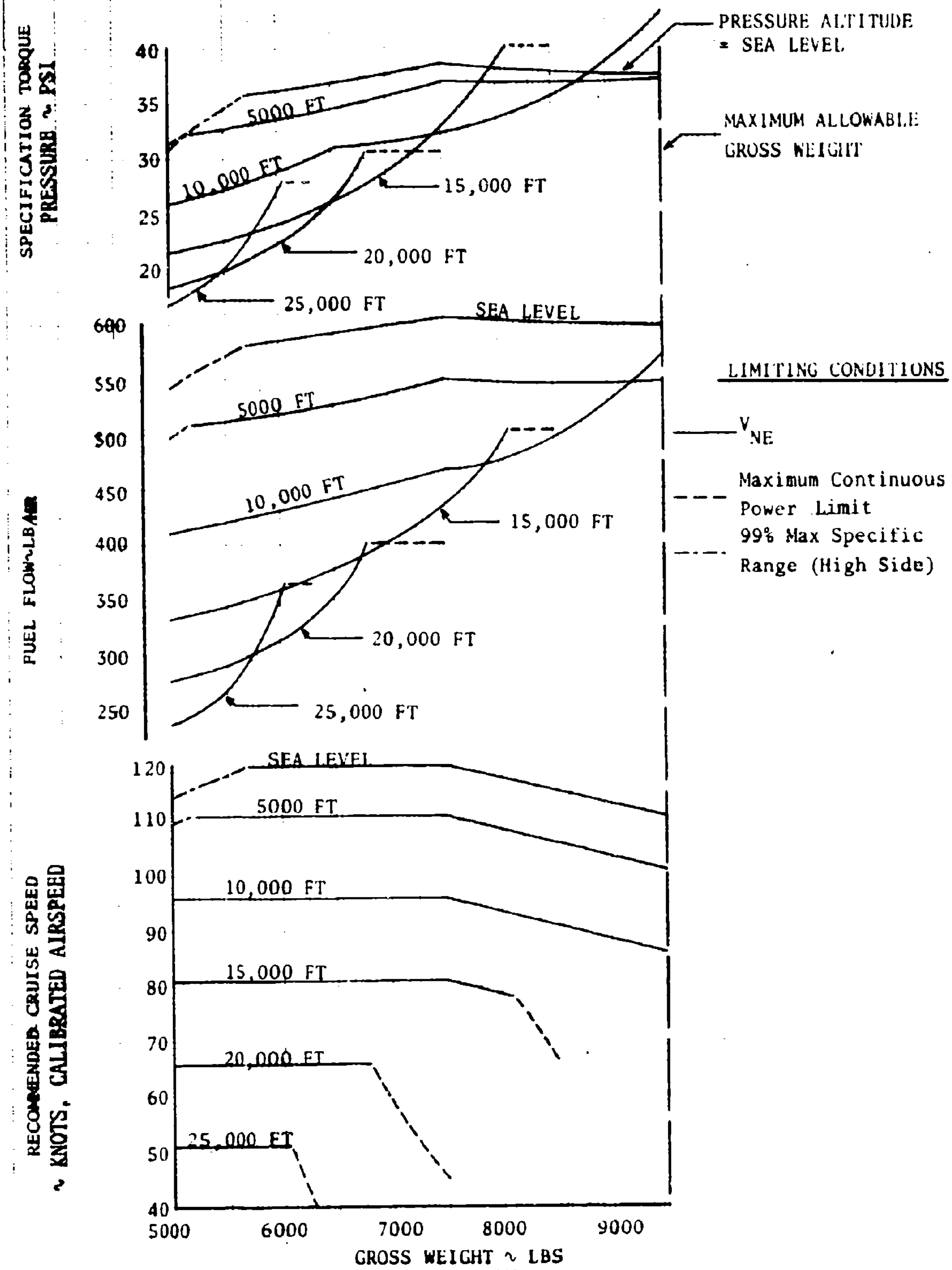


FIGURE 41

APPROXIMATE RANGE PERFORMANCE WITH EXTERNAL STORES

UH-1H T53-L-13

324 ROTOR RPM STANDARD DAY

DRAG INCREASE EQUALS 10 FT^2 EQUIVALENT FLAT PLATE AREA

NOTE: Derived from Figures 66,67,117,121,131

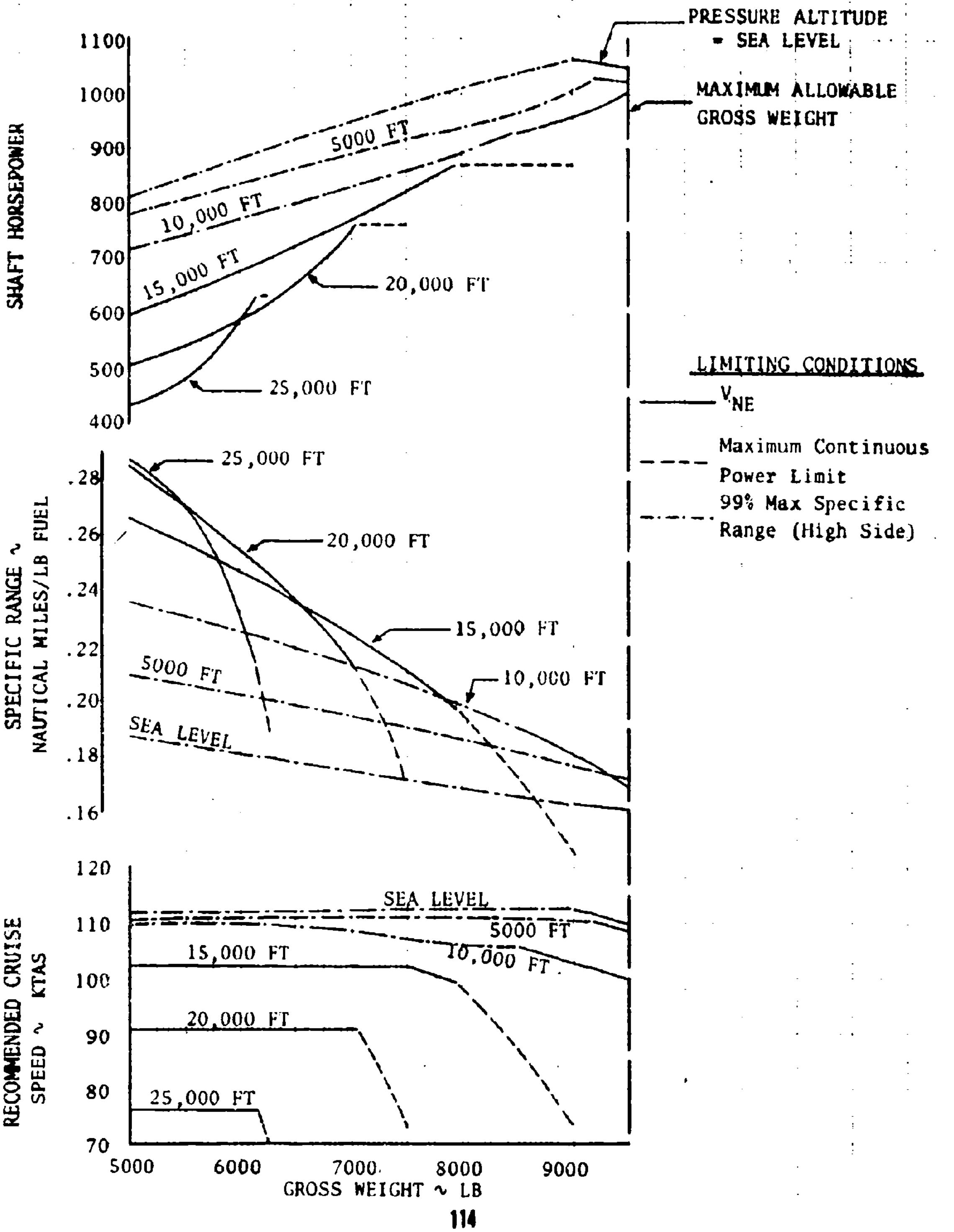


FIGURE 41 Continued
RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM STANDARD DAY

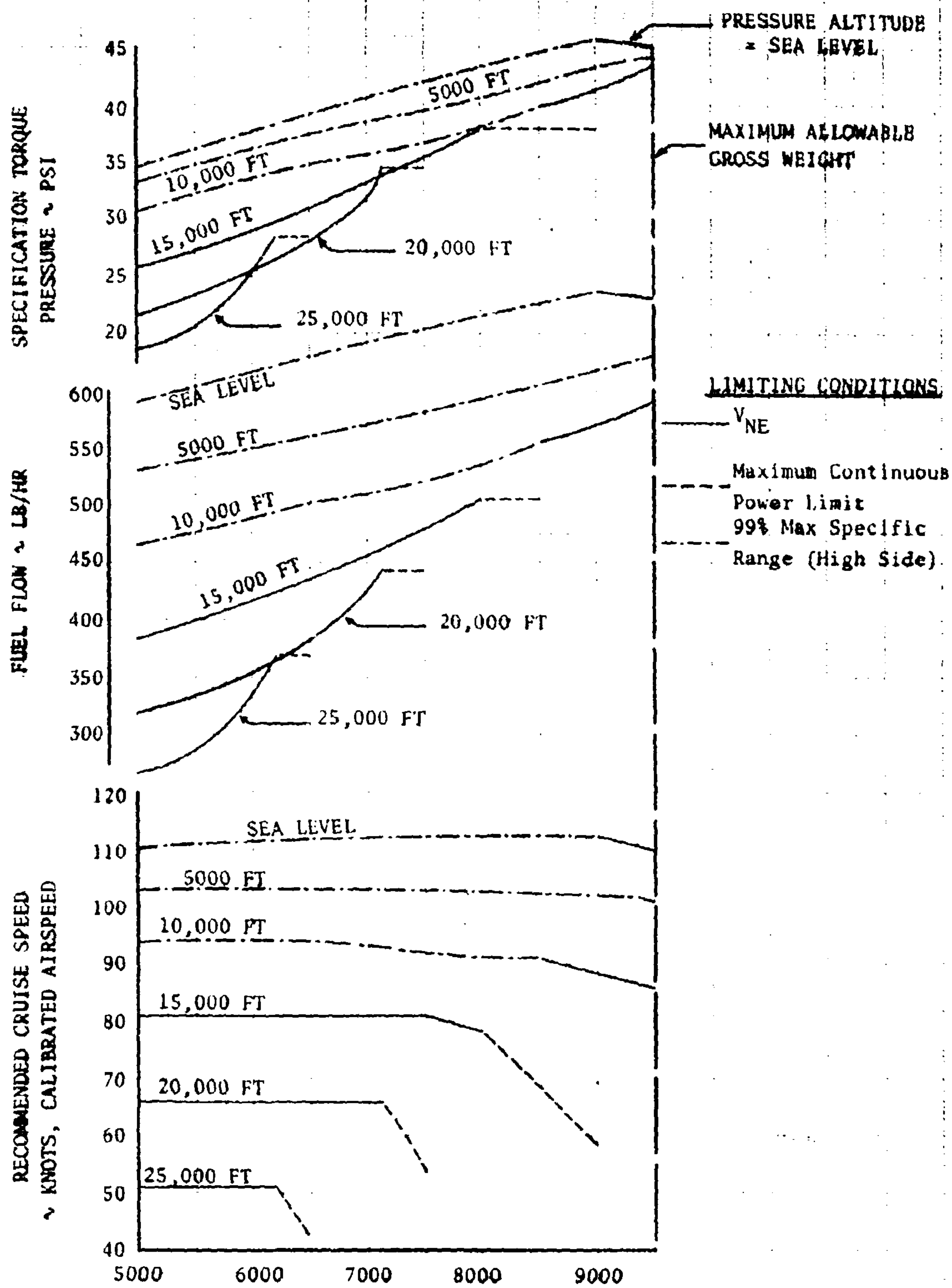


FIGURE 42

EXTRAPOLATED RANGE PERFORMANCE

UH-1H T53-L-13

324 ROTOR RPM AMBIENT TEMP = 50°C CLEAN CONFIGURATION

NOTE: Derived from Figures 66,67,117,131 and corrected specification fuel flow

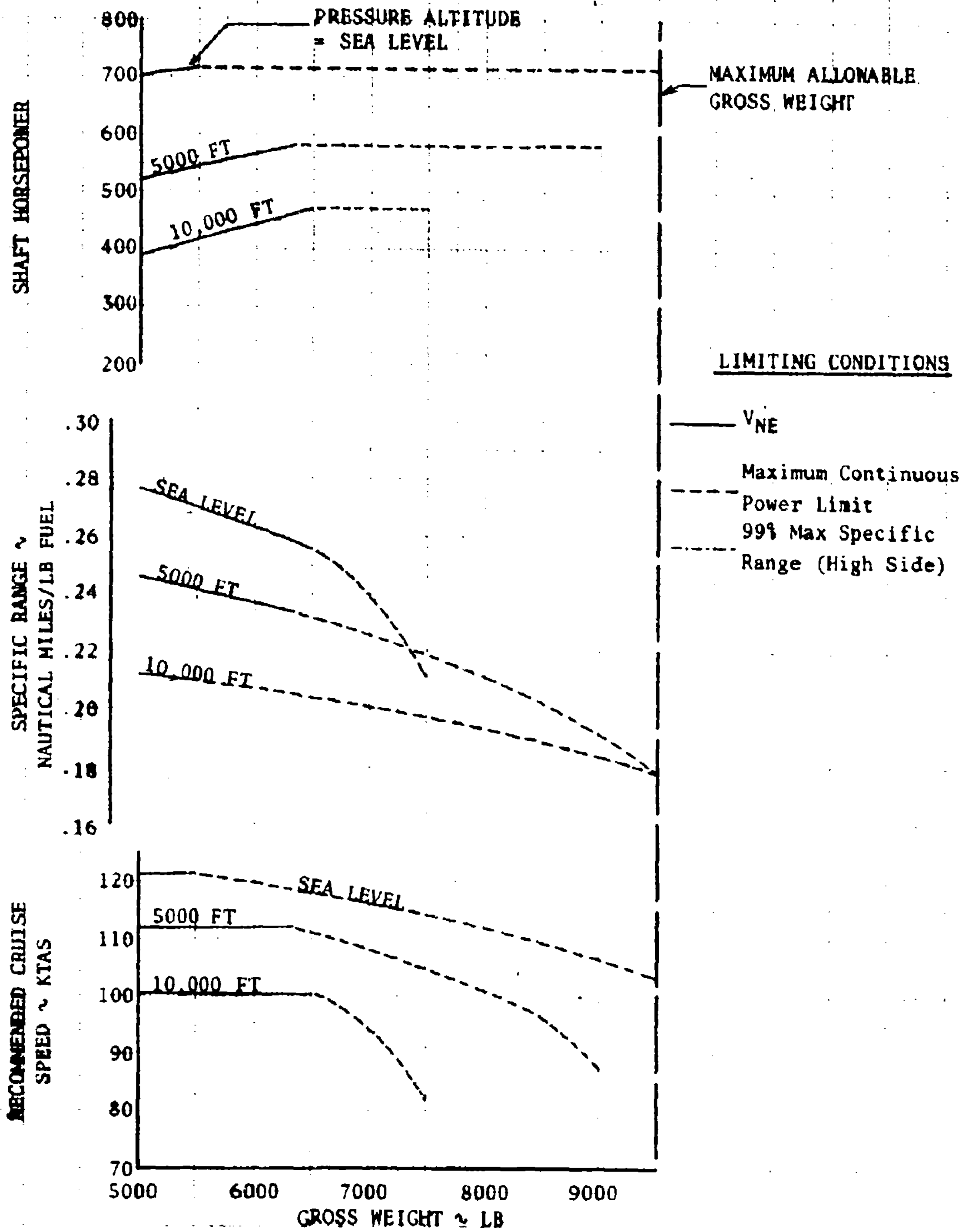


FIGURE 42 Continued
RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = 50°C

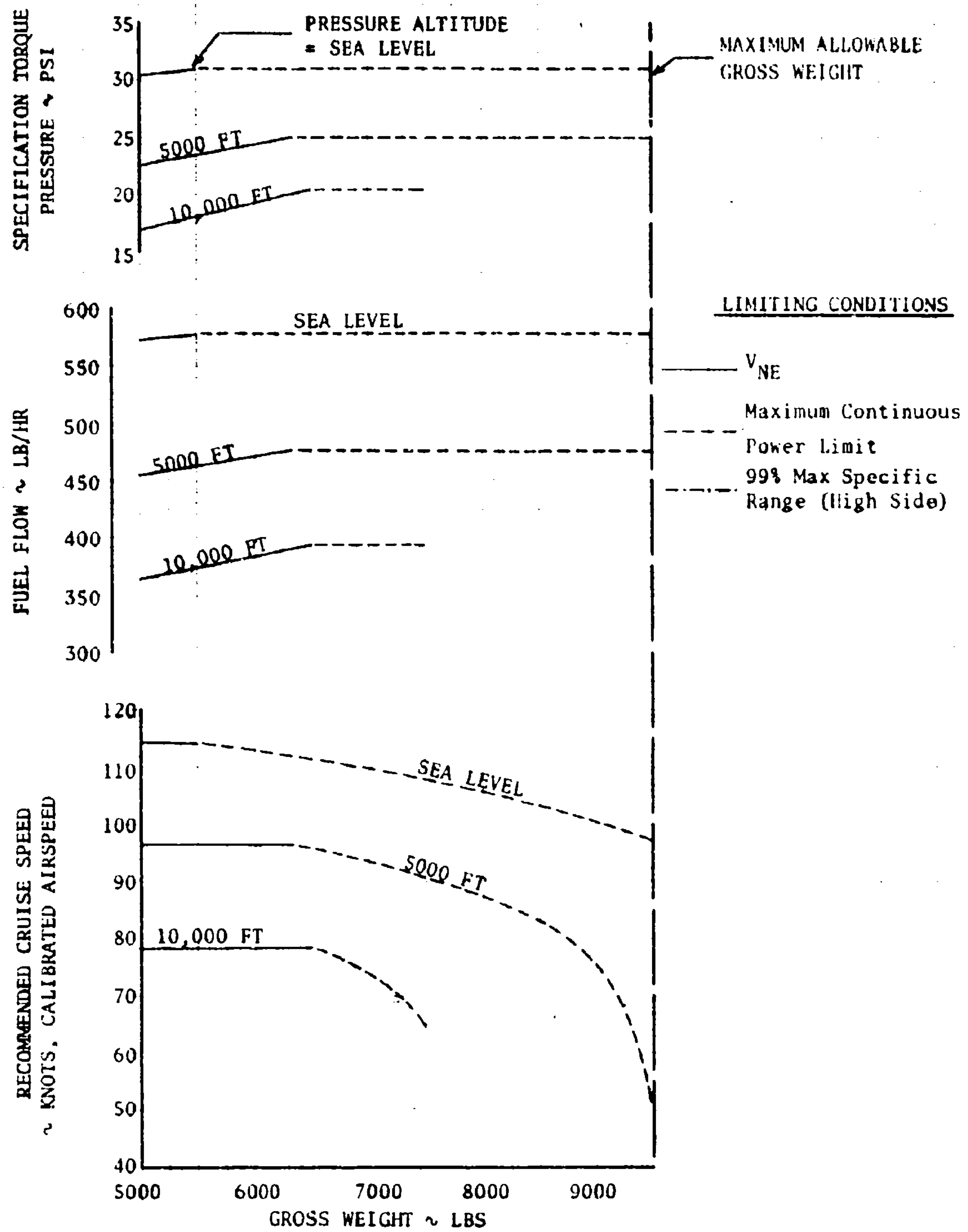


FIGURE 43
RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = 30°C CLEAN CONFIGURATION

NOTE: Derived from Figures 66,67,117,131 and corrected specification fuel flow

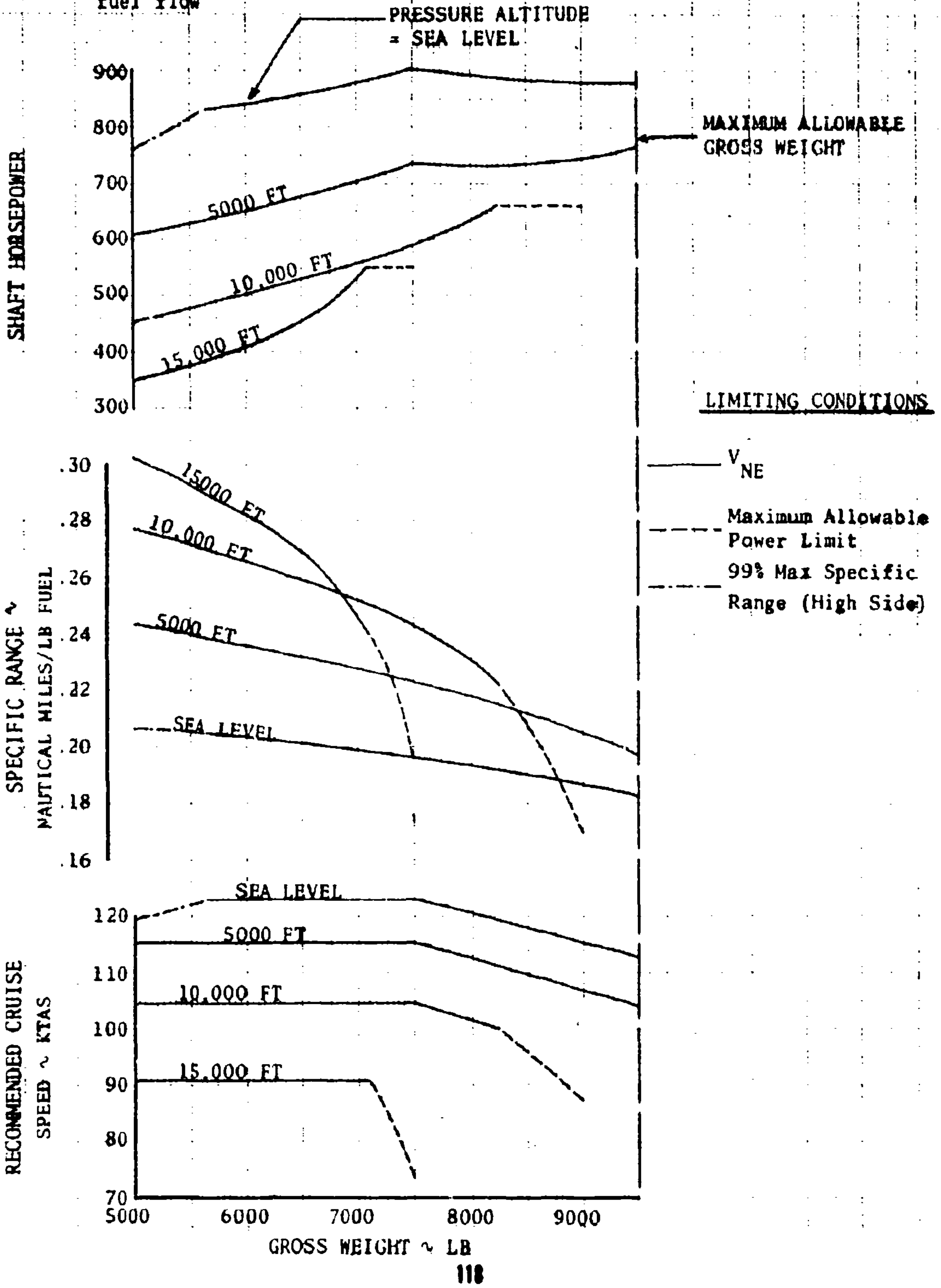


FIGURE 43 Continued
RANGE PERFORMANCE
UH-1H TS3-L-13
324 ROTOR RPM AMBIENT TEMP = 30°C

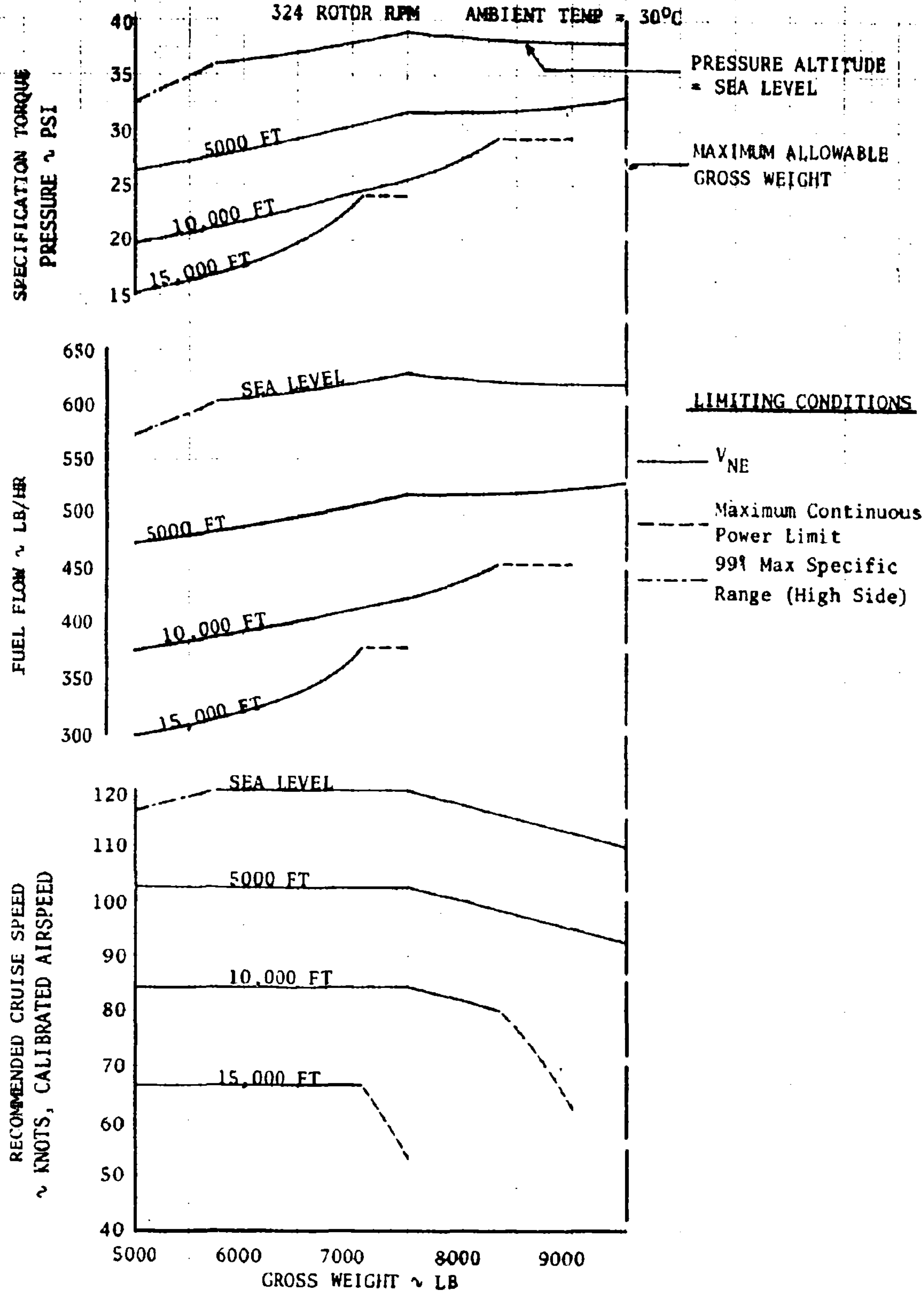


FIGURE 44
RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = 10°C CLEAN CONFIGURATION
NOTE: Derived from Figures 66,67,117,131 and corrected specification fuel flow

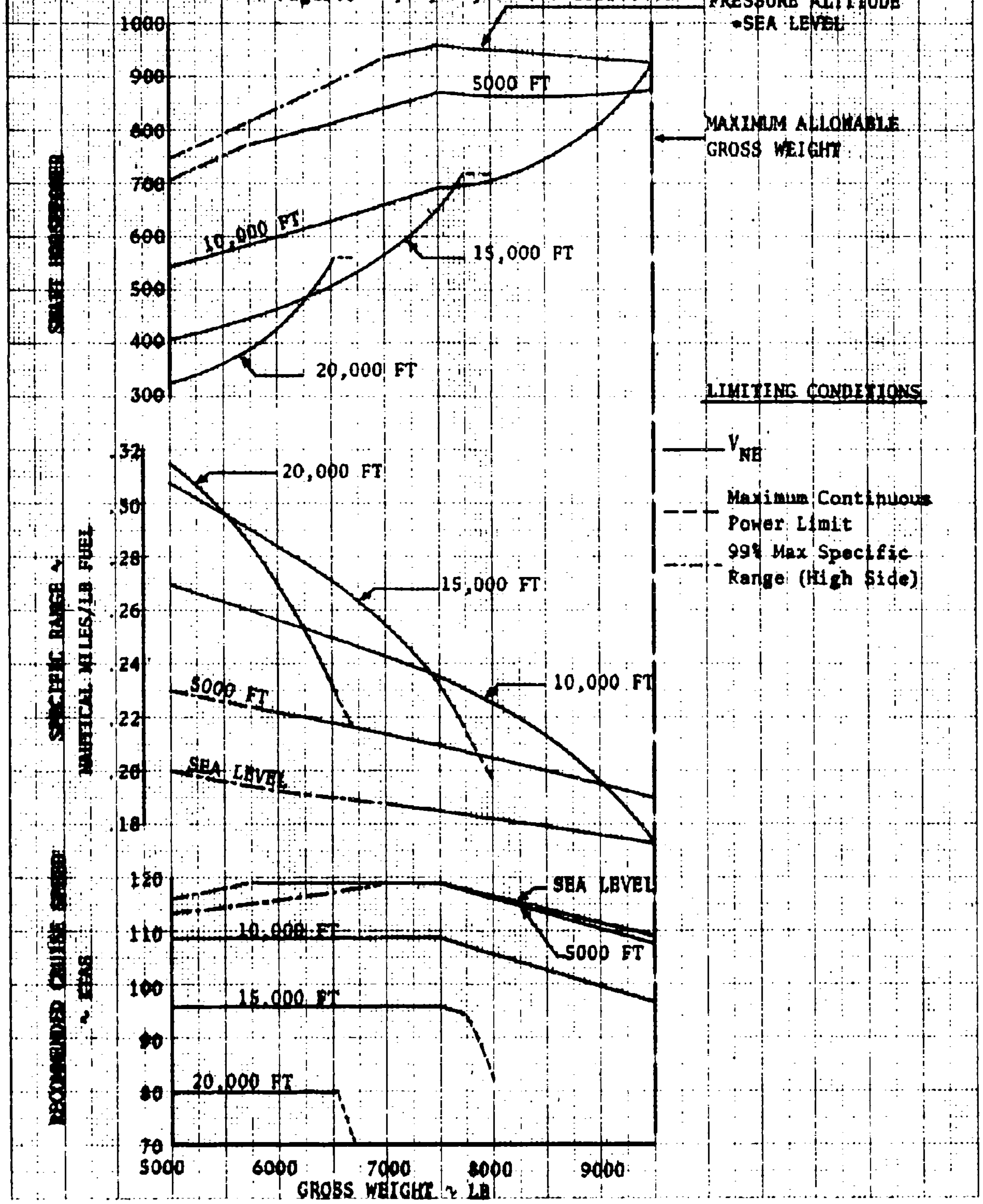


FIGURE 44 Continued
RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = 10°C

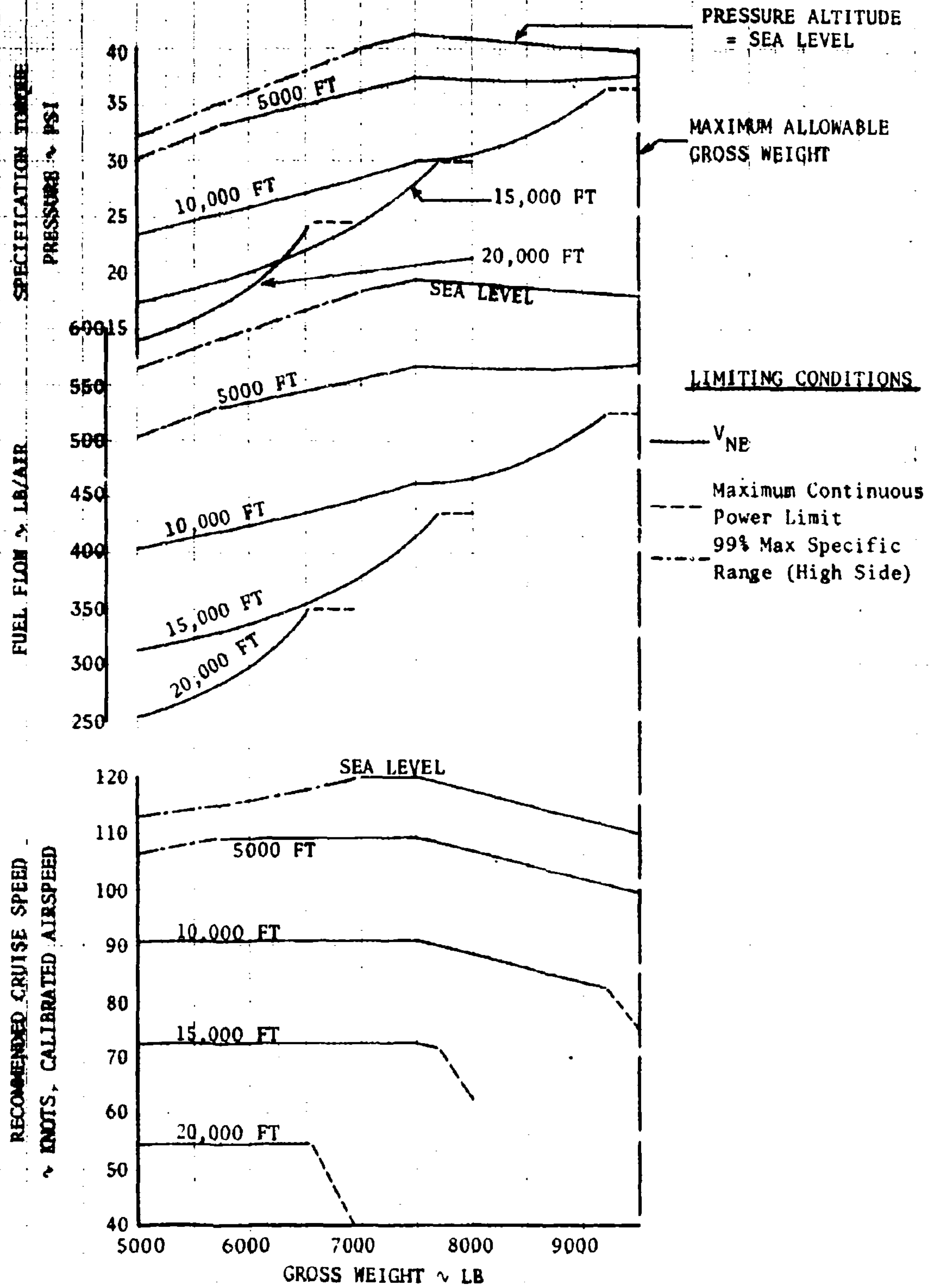


FIGURE 44
RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = -10°C CLEAN CONFIGURATION
BLEED AIR HEAT FULL ON ANTI-ICE ON OR OFF

NOTE: Derived from Figures 66,67,120,122, and 131

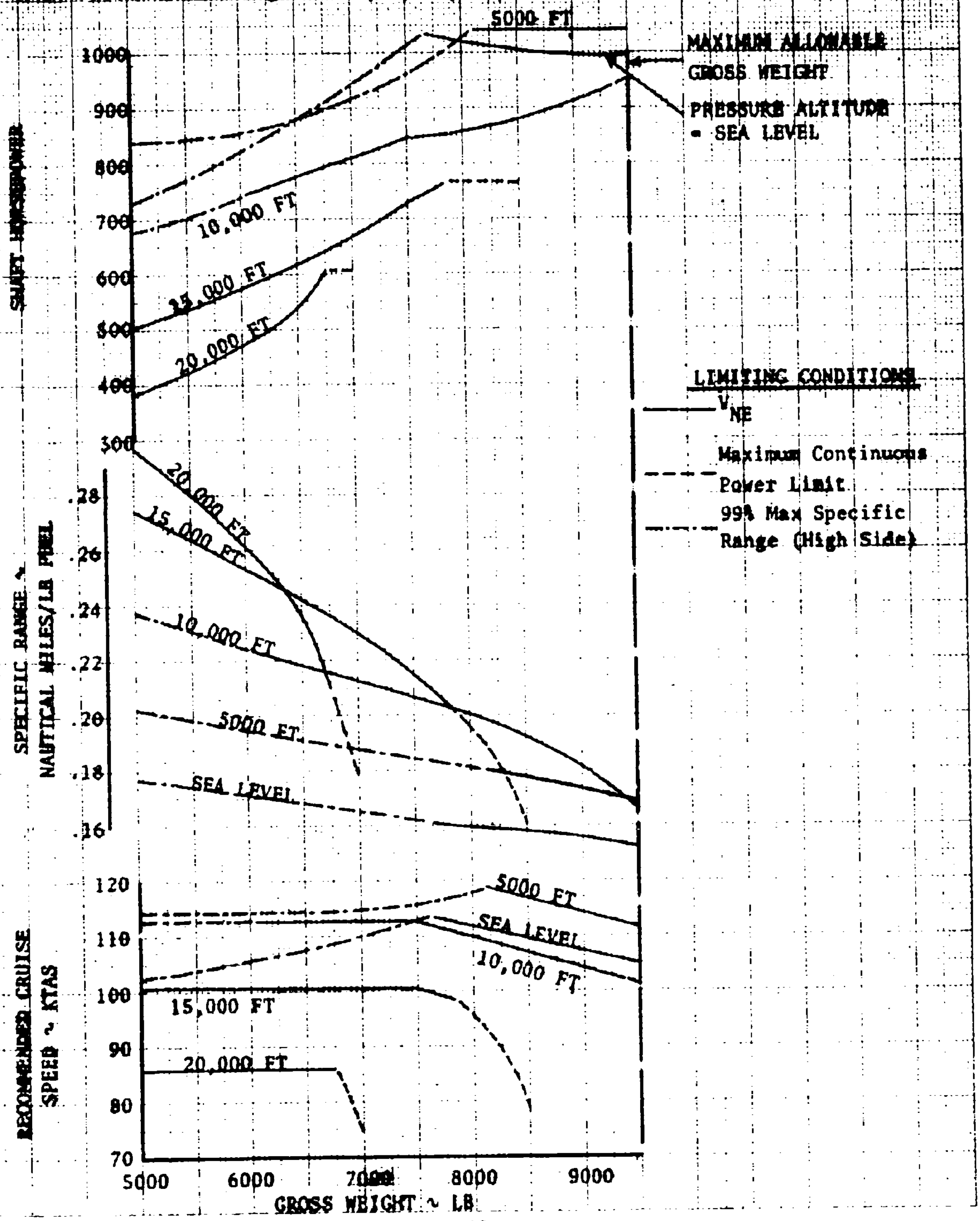


FIGURE 45 Continued

RANGE PERFORMANCE

UH-1H   T53-L-13

324 ROTOR RPM   AMBIENT TEMP = -10°C

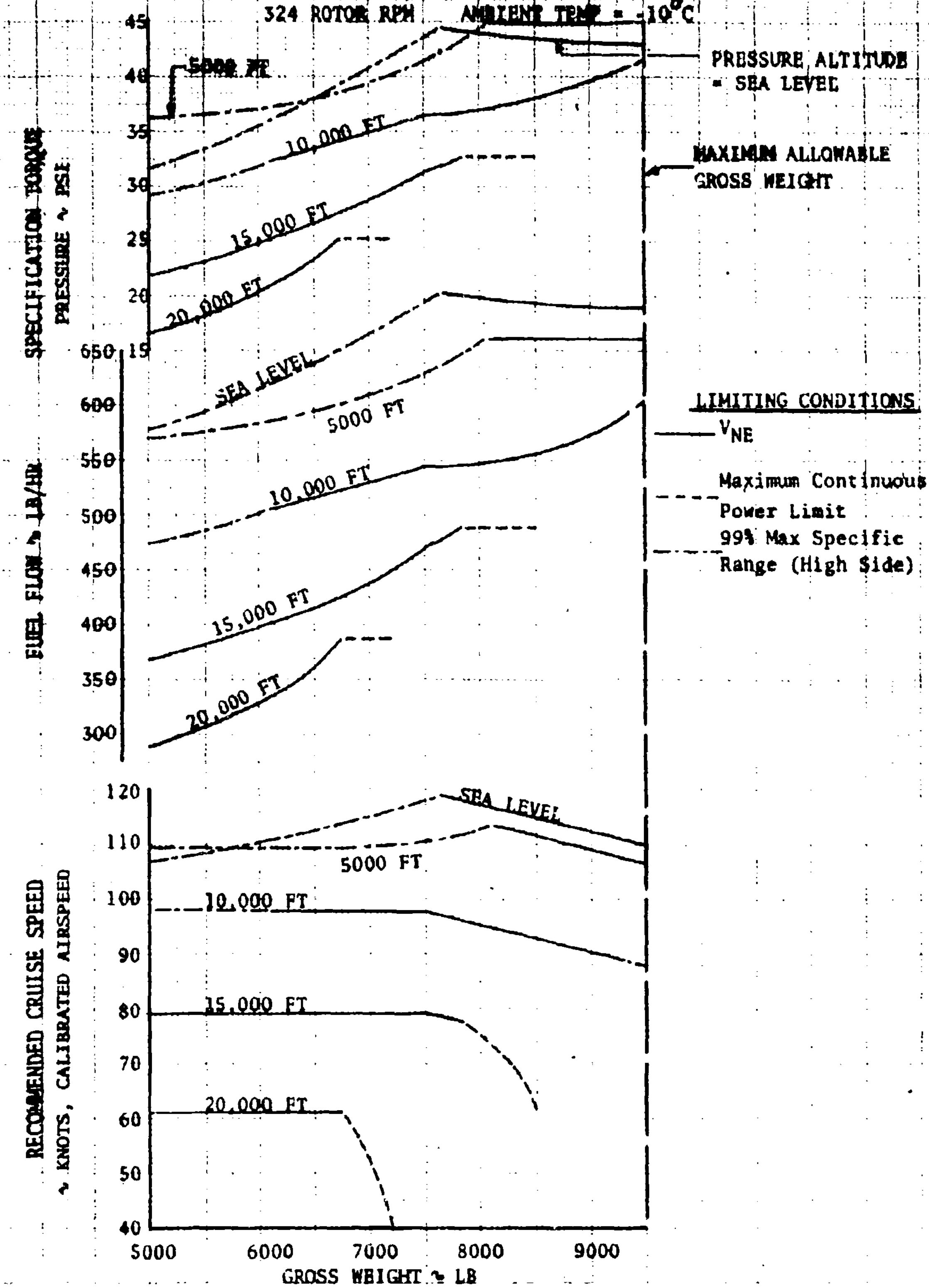


FIGURE 46
EXTRAPOLATED RANGE PERFORMANCE
UH-1M T53-L-13
324 ROTOR RPM AMBIENT TEMP = -30°C CLEAN CONFIGURATION
BLEED AIR HEAT FULL ON ANTI-ICE ON OR OFF

NOTE: Derived from Figures 66,67,120,131 and corrected specification fuel flow

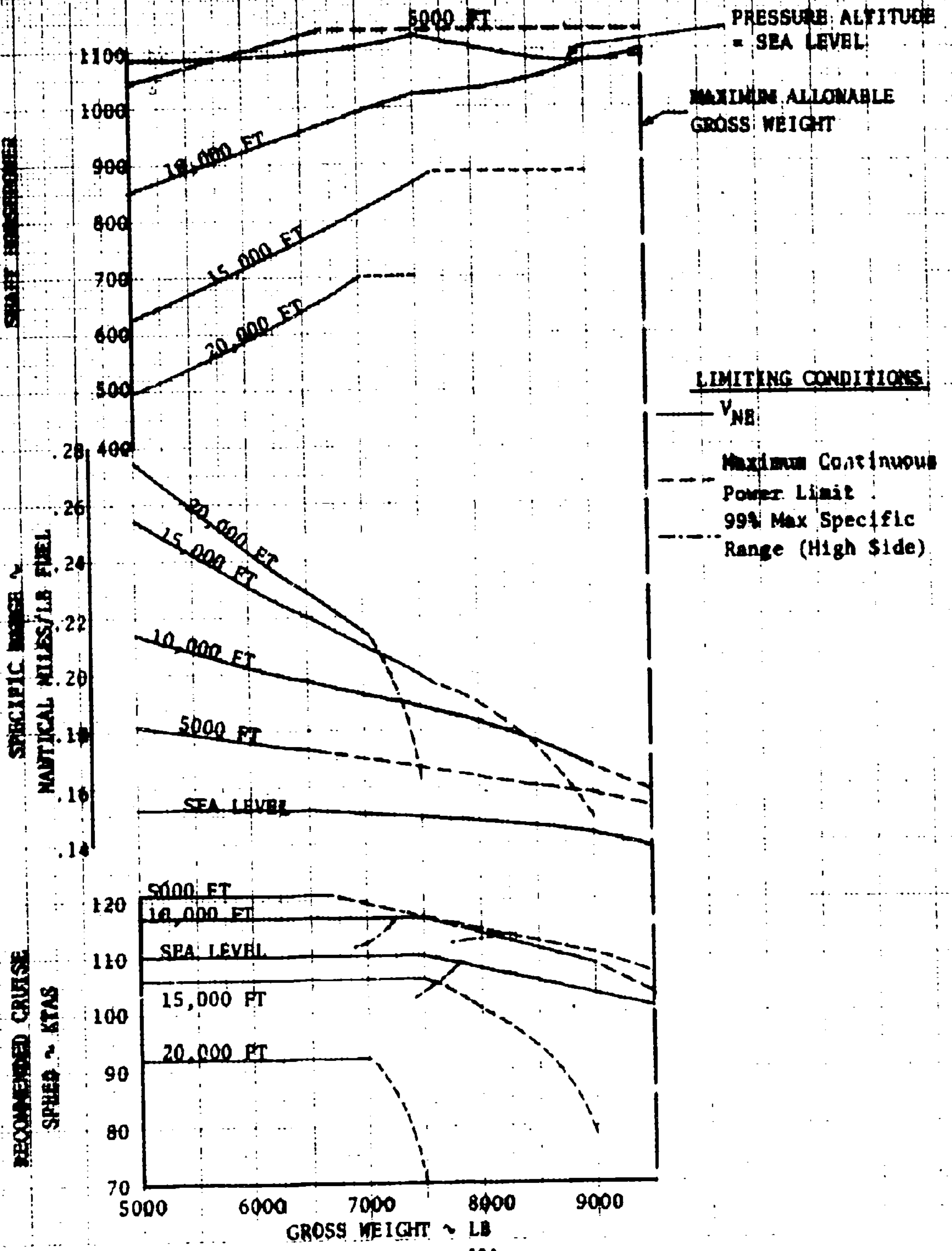


FIGURE 46 Continued
RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = -30°C

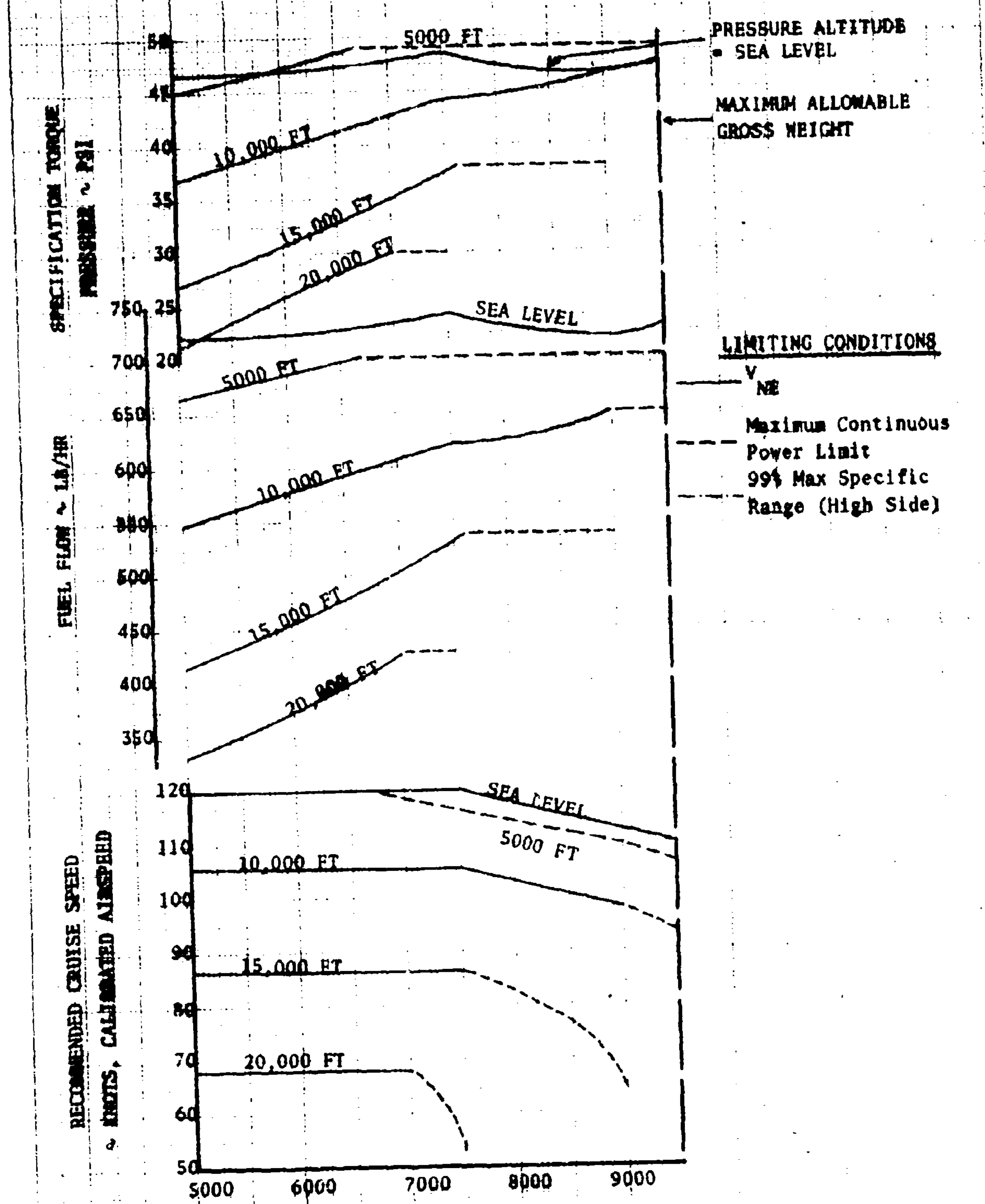


FIGURE 47
EXTRAPOLATED RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = -50°C CLEAN CONFIGURATION
BLEED AIR HEAT FULL ON ANTI-ICE ON OR OFF
NOTE: Derived from Figures 44,47,120,131 and corrected specification fuel flow

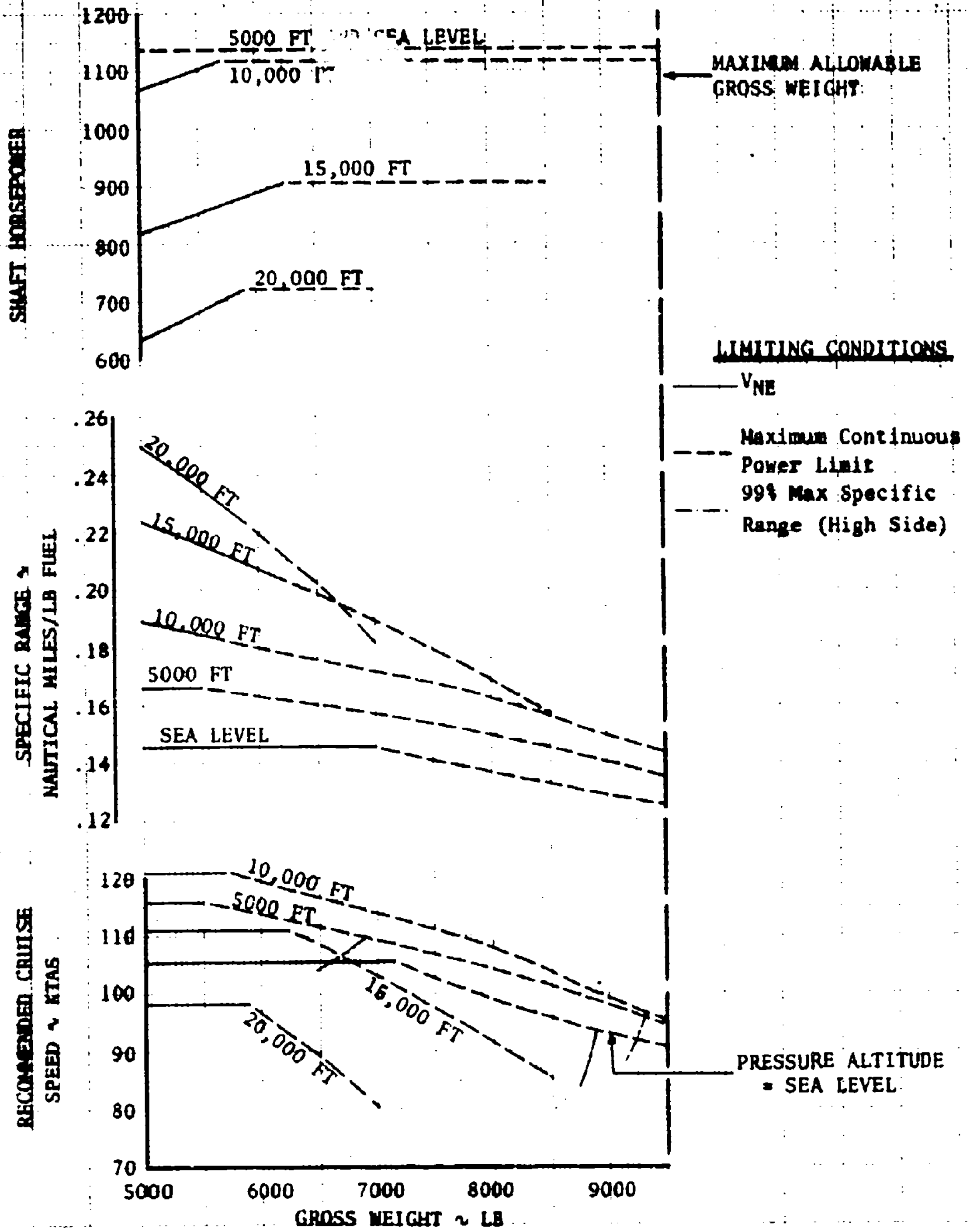


FIGURE 47 Continued
RANGE PERFORMANCE
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = -50°C

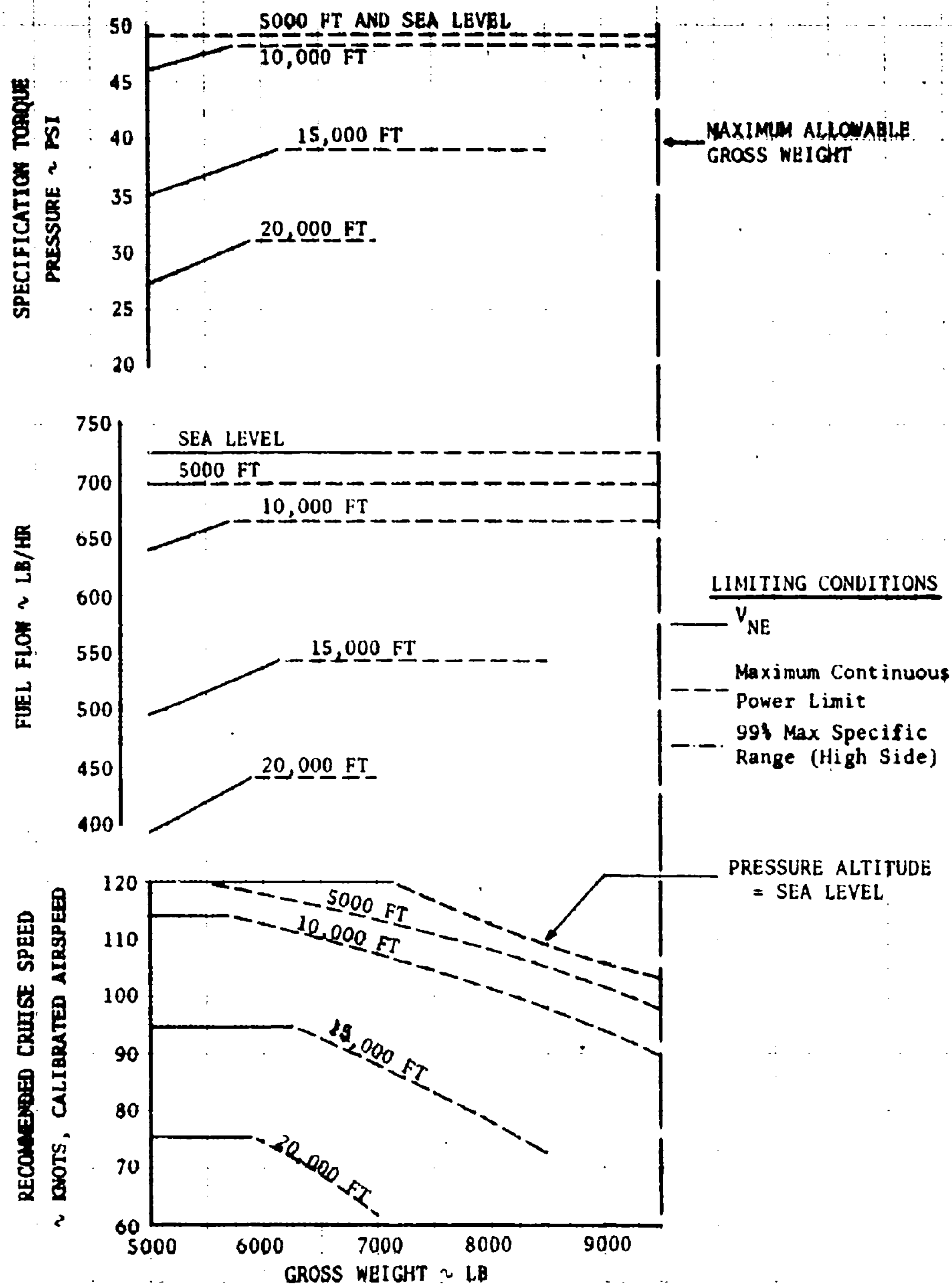


FIGURE 48
ENDURANCE SUMMARY
UH-1H T53-L-13
324 ROTOR RPM STANDARD DAY

NOTE: Derived From Figures 66 , and, 67 , and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 117

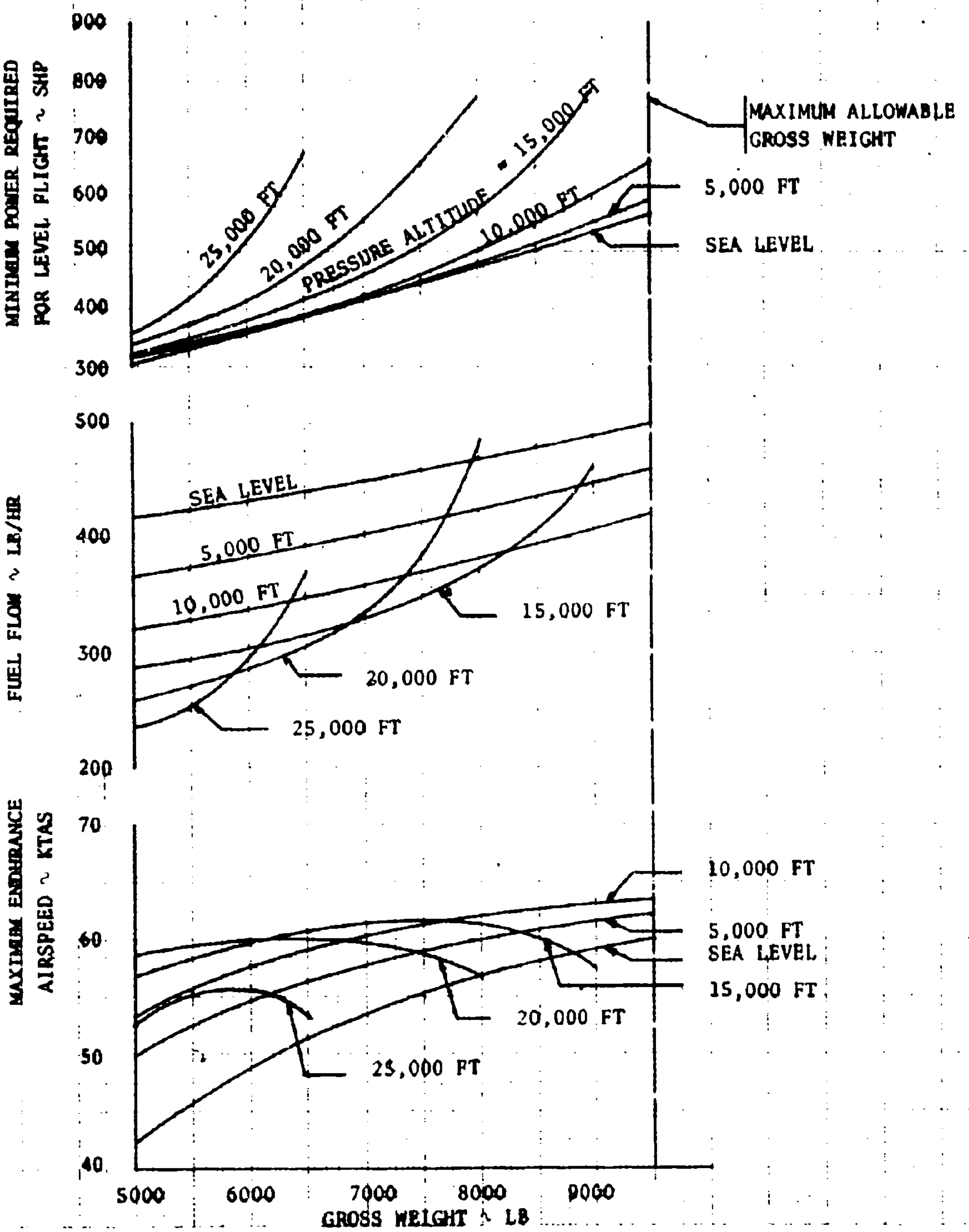


FIGURE 49
ENDURANCE SUMMARY
UH-1H T53-L-13
314 ROTOR RPM STANDARD DAY

NOTE: Derived From Figures 66, and, 67 and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 117

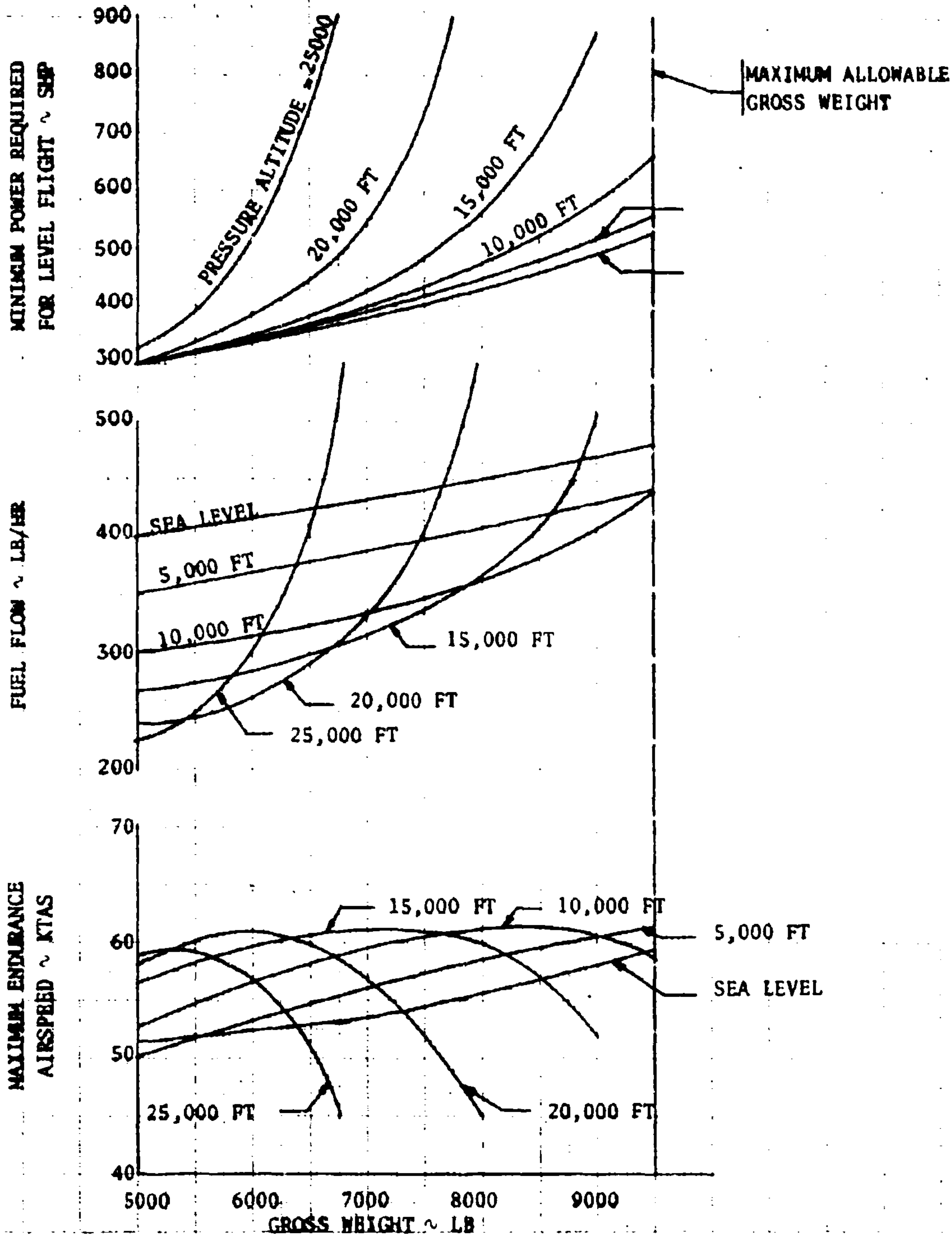


FIGURE 50
ENDURANCE SUMMARY
UH-1M T53-L-13
294 ROTOR RPM STANDARD DAY

NOTE: Derived From Figures 66 , and, 67 , and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 117

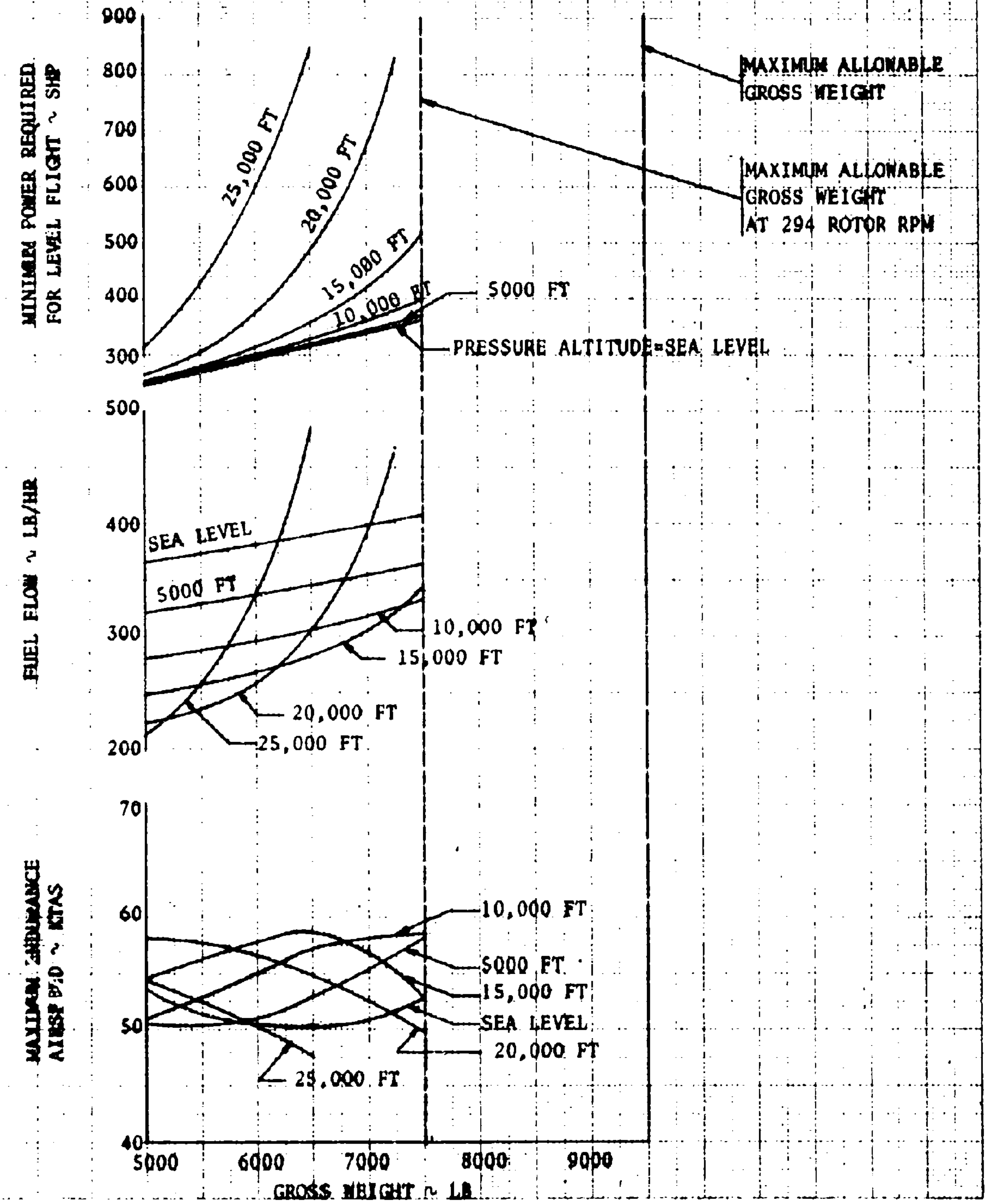


FIGURE 51
EXTRAPOLATED ENDURANCE SUMMARY
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = 50°C

NOTE: Derived From Figures 66 , and, 67 and Specification Fuel Flow Corrected for Installation Losses Listed on Figure 117

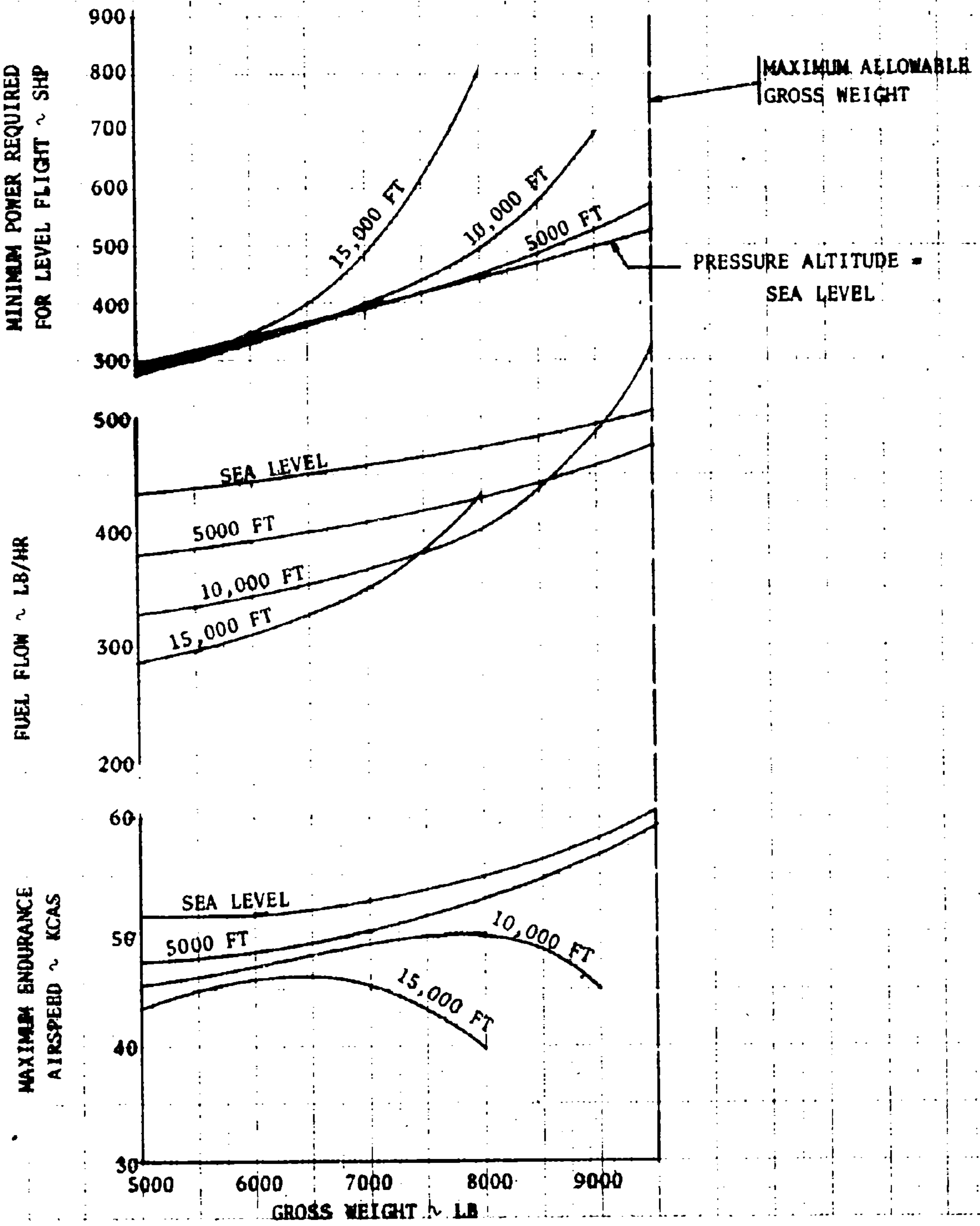


FIGURE 52
ENDURANCE SUMMARY
UH-1H TS3-L-13
324 ROTOR RPM AMBIENT TEMP = 30°C

NOTE: Derived From Figures 66 , and, 67 and Specification Fuel Flow Corrected for Installation Losses Listed on Figure 117

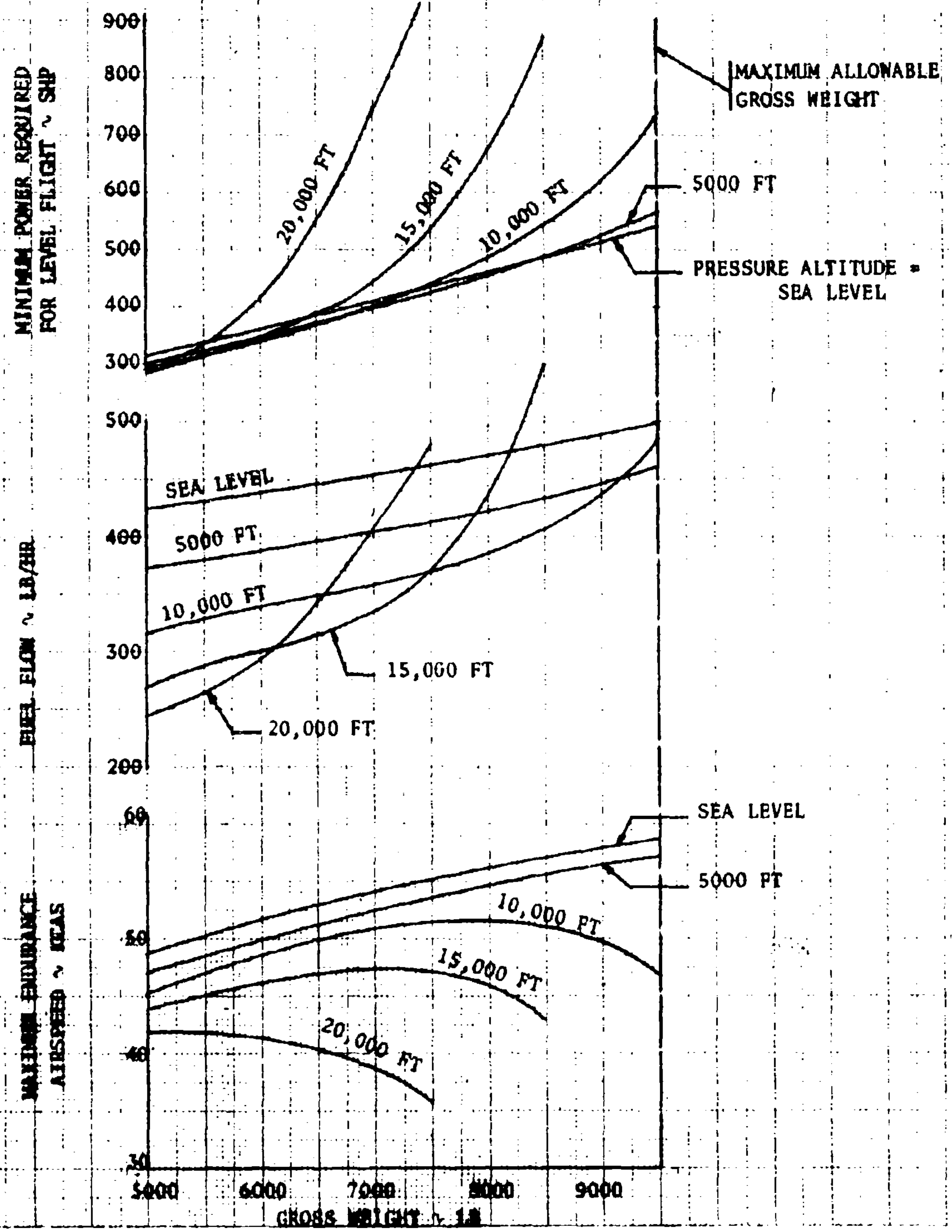


FIGURE 53
ENDURANCE SUMMARY
UH-1H T53-L-13
324 ROTOR RPM AMBIENT TEMP = 10°C

NOTE: Derived From Figures 66., and, 67 and Specification Fuel Flow Corrected for Installation Losses Listed on Figure 117

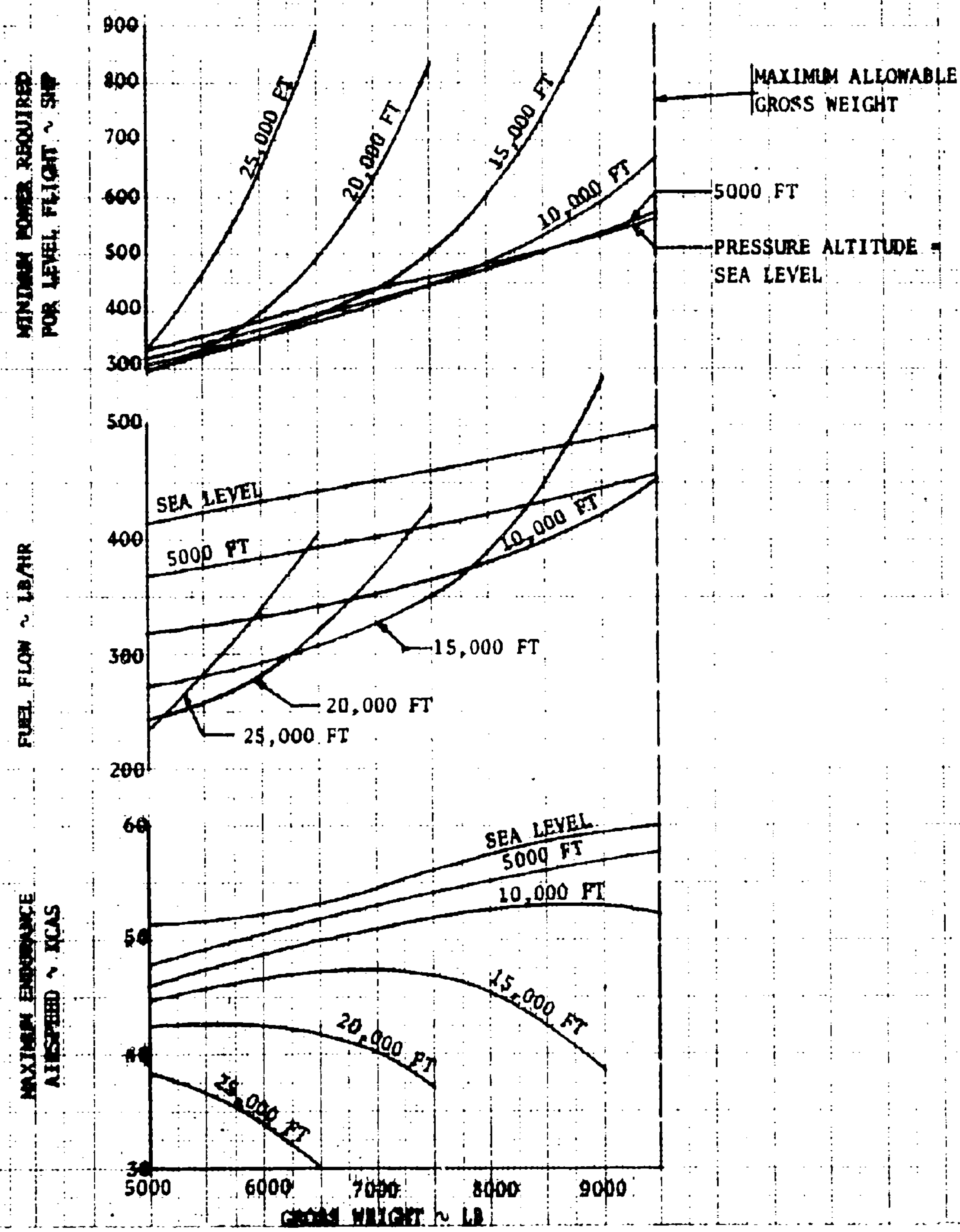


FIGURE 54

ENDURANCE SUMMARY

UH-1H T53-L-13

324 ROTOR RPM AMBIENT TEMP = -10°C

BLEED AIR HEAT FUEL ON ANTI-ICE ON OR OFF

NOTE: Derived From Figures 66 , and, 67 and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 120

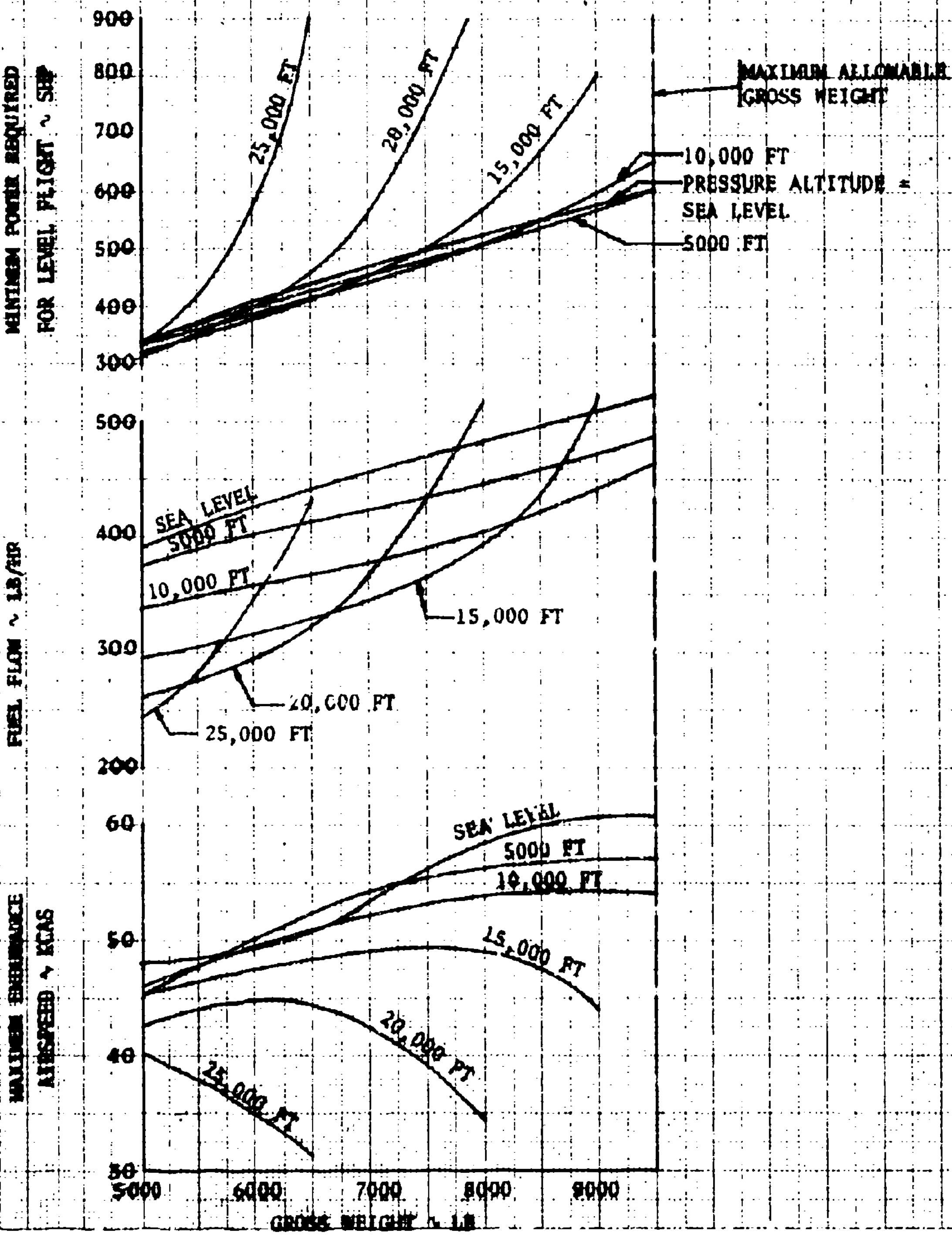


FIGURE 55

EXTRAPOLATED ENDURANCE SUMMARY

UH-1H T53-L-13

324 ROTOR RPM AMBIENT TEMP = -30°C

BLEED AIR HEAT FULL ON ANTI-ICE ON OR OFF

NOTE: Derived From Figures 66 , and, 67 and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 120

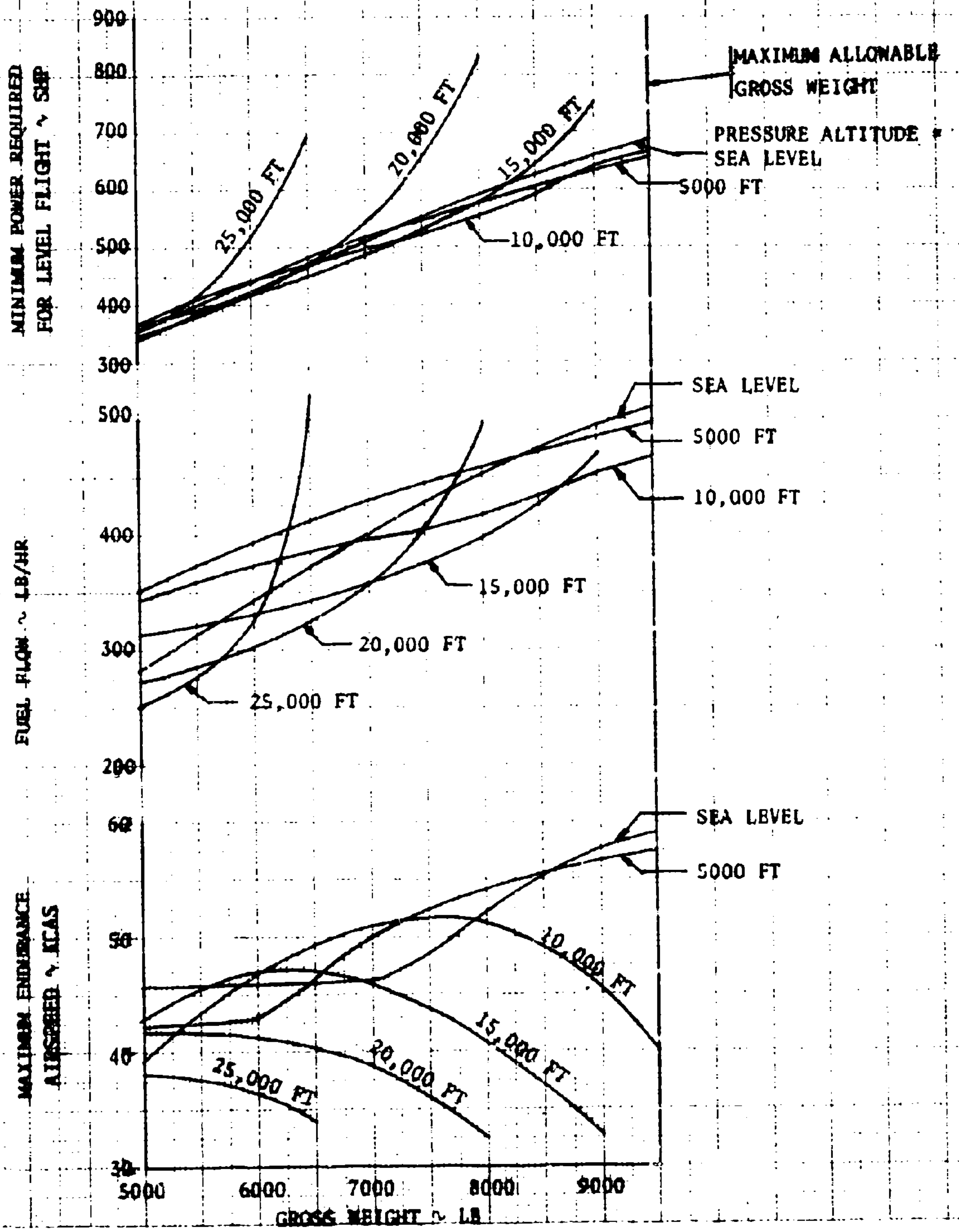


FIGURE 56

EXTRAPOLATED ENDURANCE SUMMARY

UH-1H   T53-L-13

324 ROTOR RPM   AMBIENT TEMP = -50°C

BLEED AIR HEAT FULL ON   ANTI-ICE ON OR OFF

NOTE: Derived From Figures 66 , and, 67 and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 120

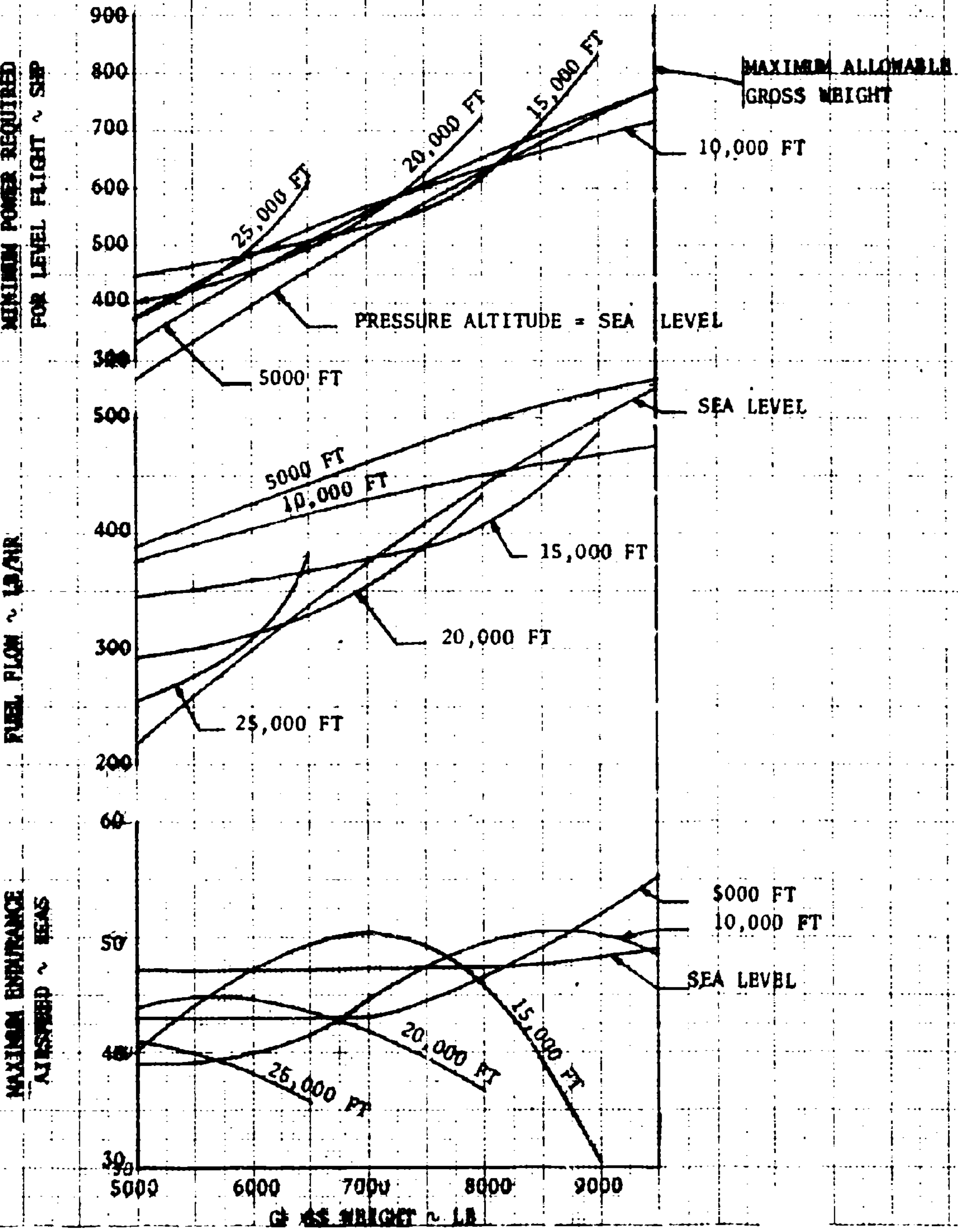


FIGURE 57
ENDURANCE SUMMARY
UH-1H T53-L-13

AIRSPEED = 50 KCAS ROTOR SPEED = 324 RPM

STANDARD DAY

NOTE: Derived from Figures 66 and 67, and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 117

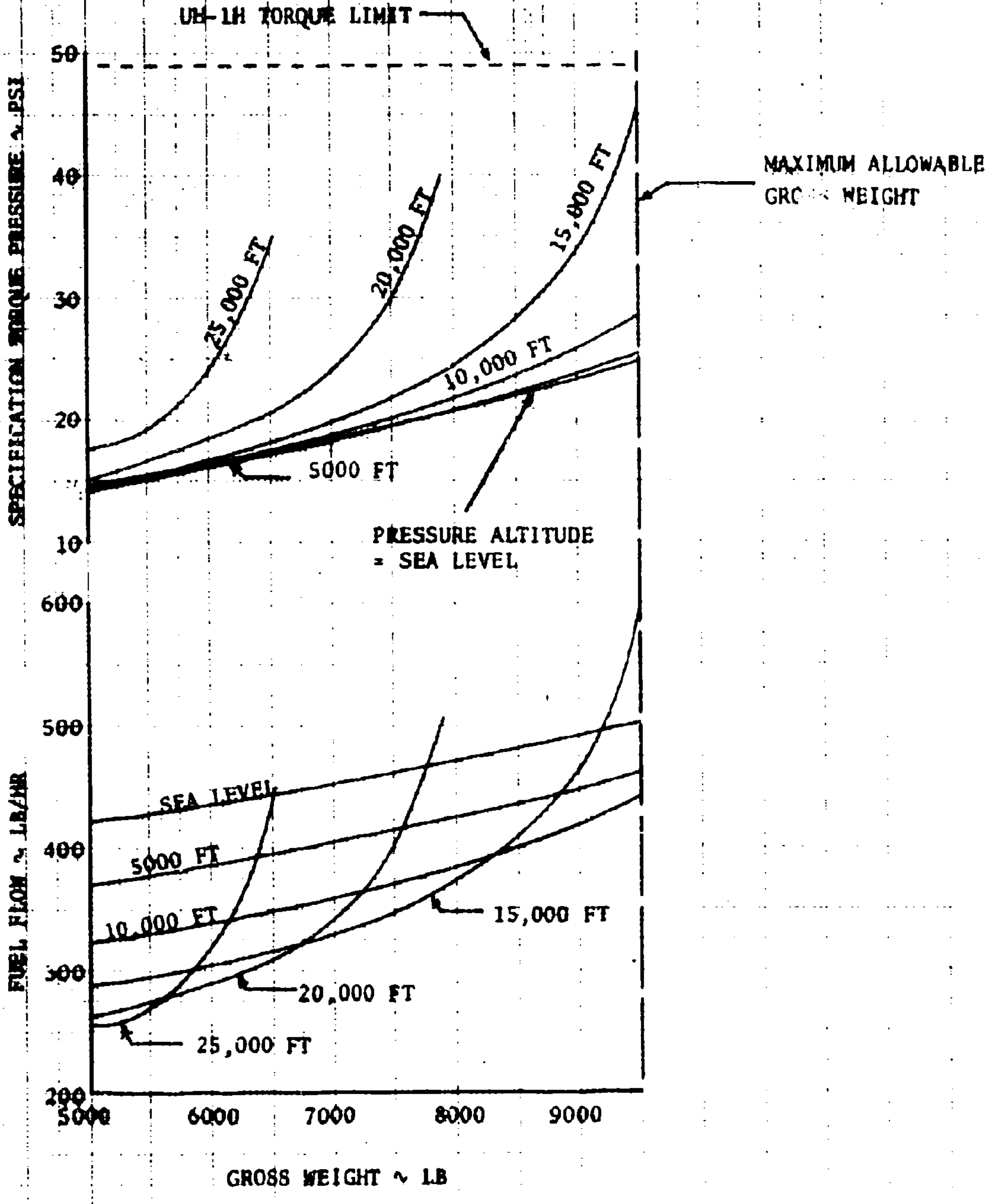


FIGURE 58
ENDURANCE SUMMARY
UH-1H T53-L-13

AIRSPEED = 50 KCAS ROTOR SPEED = 314 RPM
STANDARD DAY

NOTE: Derived from Figures 66 and 67, and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 117

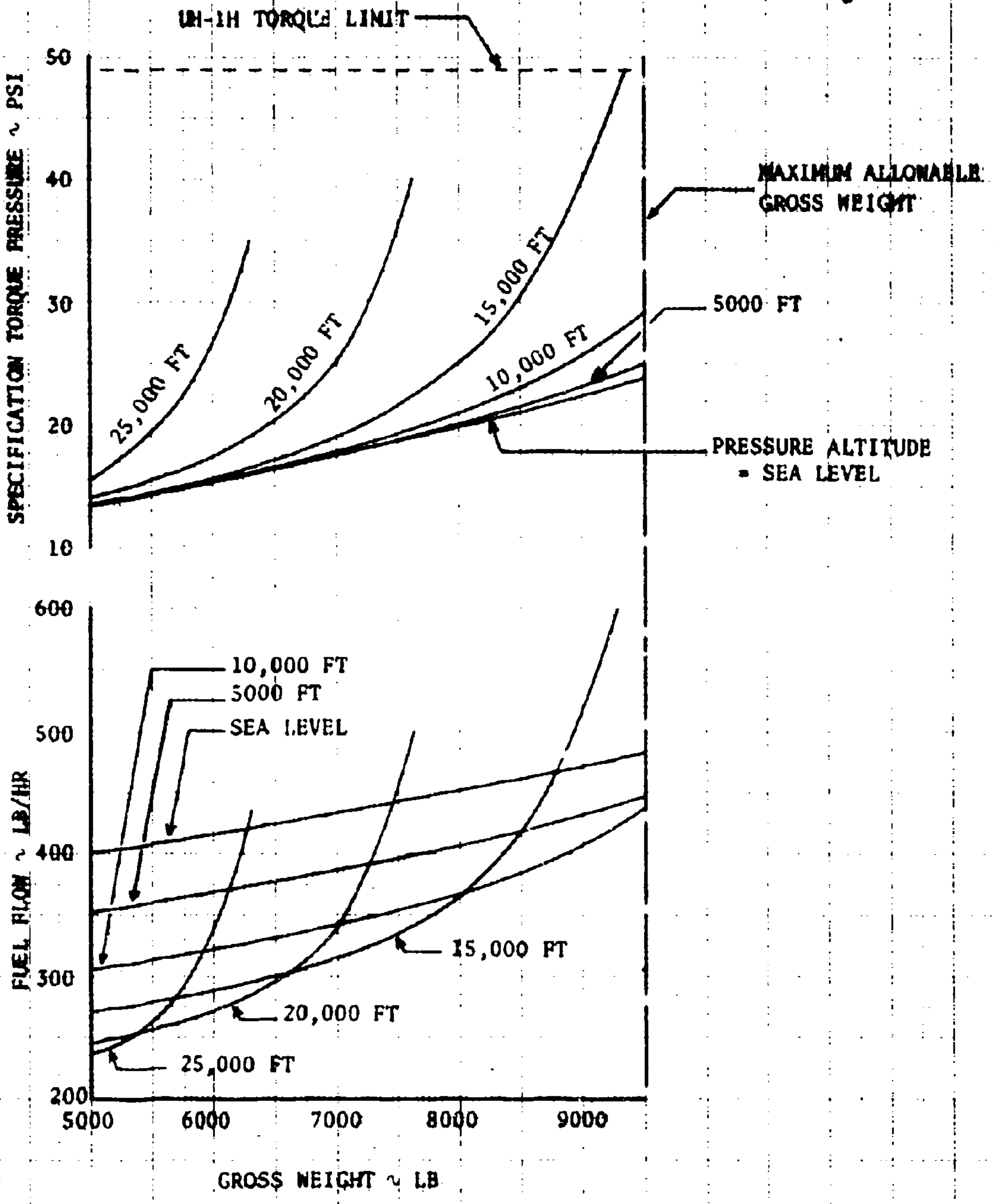


FIGURE 59
ENDURANCE SUMMARY
UH-1H T53-L-13

AIRSPEED = 50 KCAS ROTOR SPEED = 294 RPM

STANDARD DAY

NOTE: Derived from Figures 66 and 67, and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 117

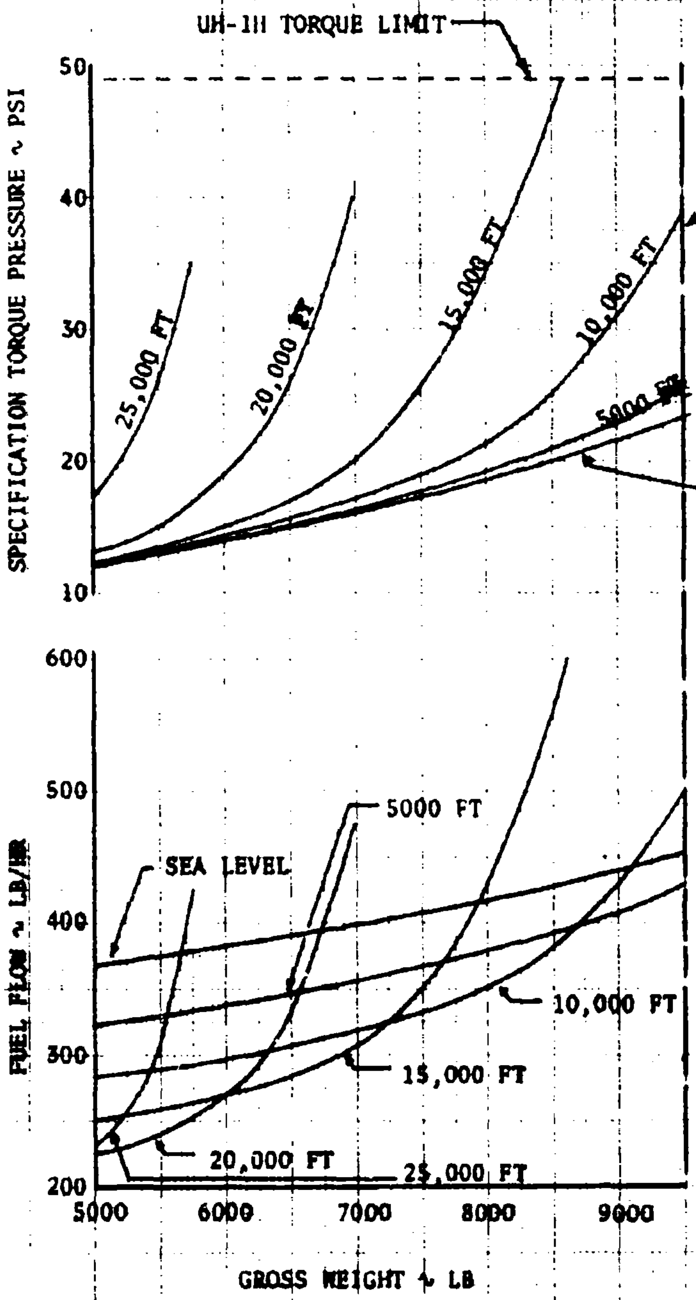


FIGURE 60
EXTRAPOLATED ENDURANCE SUMMARY
UH-1H T53-L-13

AIRSPEED = 50 KCAS ROTOR SPEED = 324 RPM
AMBIENT TEMPERATURE = 50°C

NOTE: Derived from Figures 66 and 67, and Specification Fuel Flow Corrected for Installation Losses Listed on Figure 117

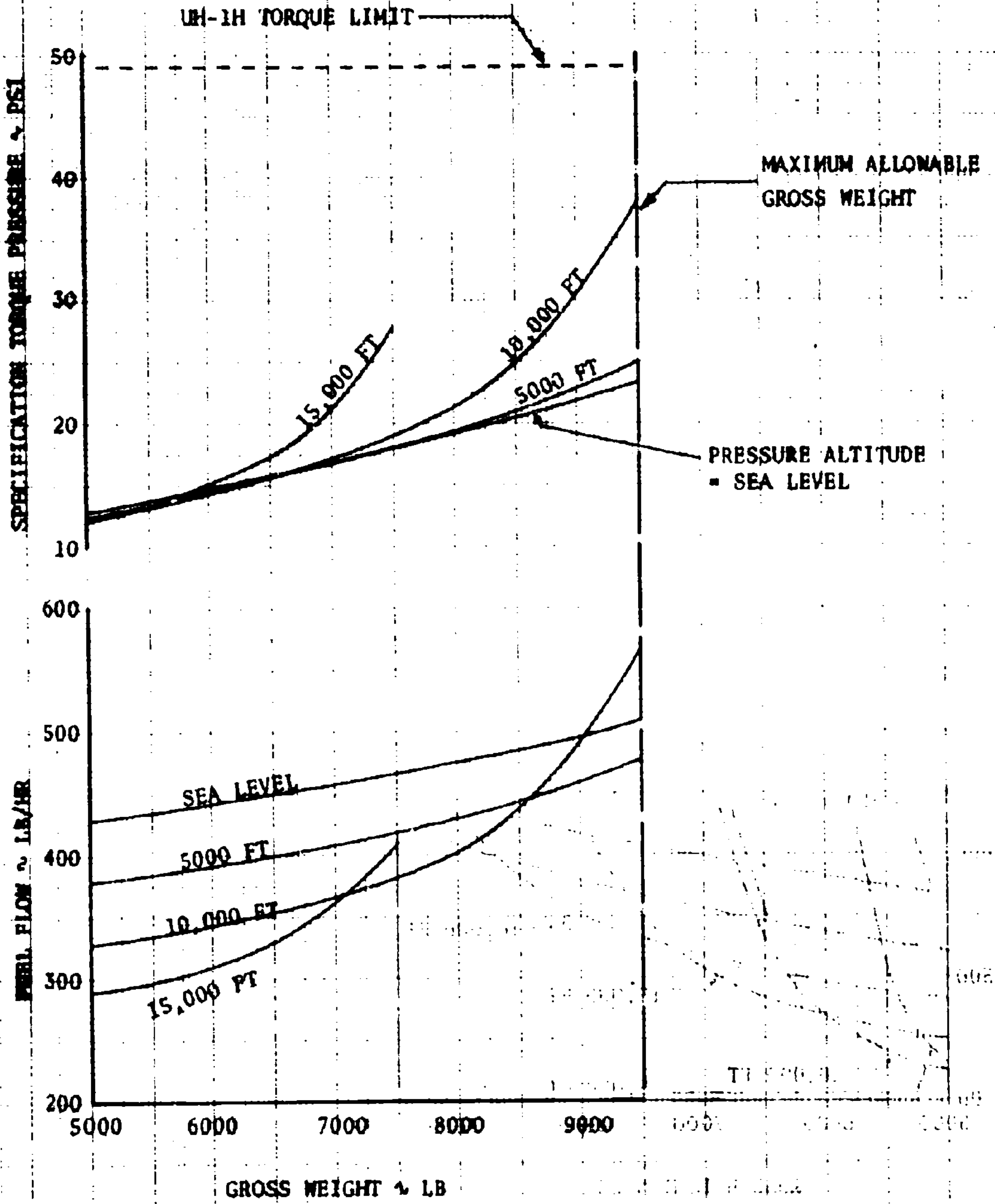


FIGURE 61
ENDURANCE SUMMARY
UH-1H T53-L-13

AIRSPEED = 50 KCAS ROTOR SPEED = 324 RPM
AMBIENT TEMPERATURE = 30°C

NOTE: Derived from Figures 66 and 67, and Specification Fuel Flow Corrected for Installation Losses Listed on Figure 117

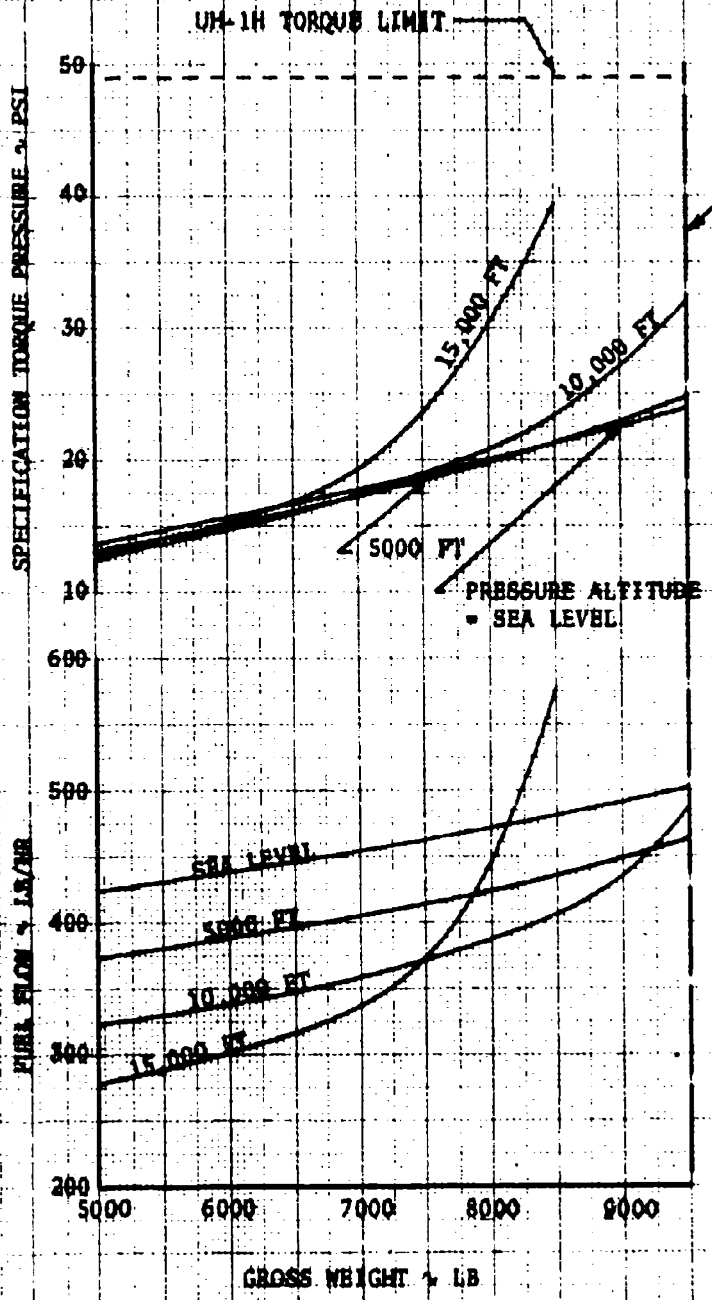


FIGURE 62
ENDURANCE SUMMARY
UH-1M T53-L-13

AIRSPEED = 50 KCAS

ROTOR SPEED = 324 RPM

AMBIENT TEMPERATURE = 10°C

NOTE: Derived from Figures 66 and 67, and Specification Fuel Flow Corrected for Installation Losses Listed on Figure 117

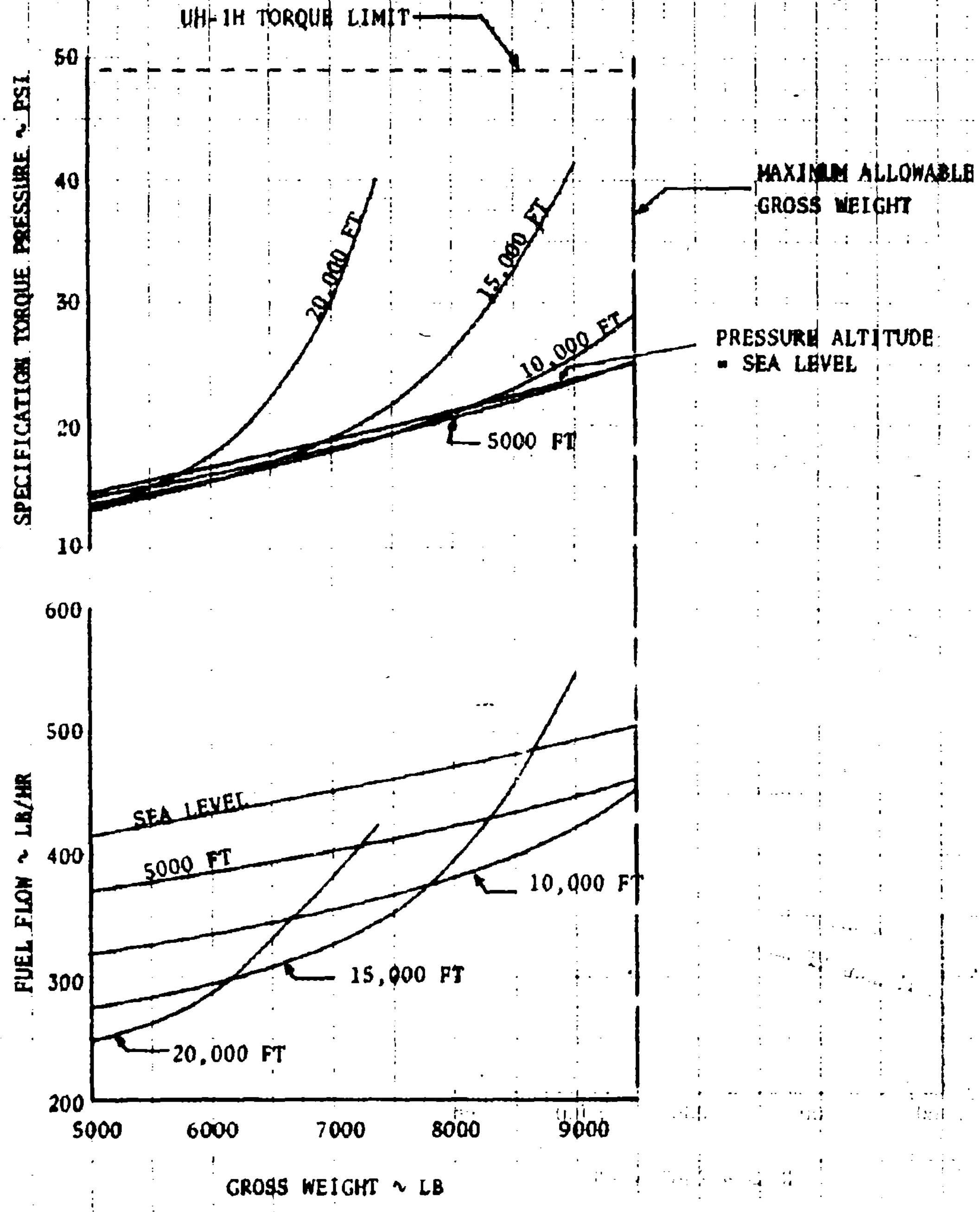


FIGURE 63
ENDURANCE SUMMARY
UH-1H T53-L-13

AIRSPEED = 50 KCAS ROTOR SPEED = 324 RPM
AMBIENT TEMPERATURE = $-10^{\circ}C$
BLEED AIR HEAT FULL ON
ANTI ICE ON OR OFF

NOTE: Derived from Figures 66 and 67, and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 120

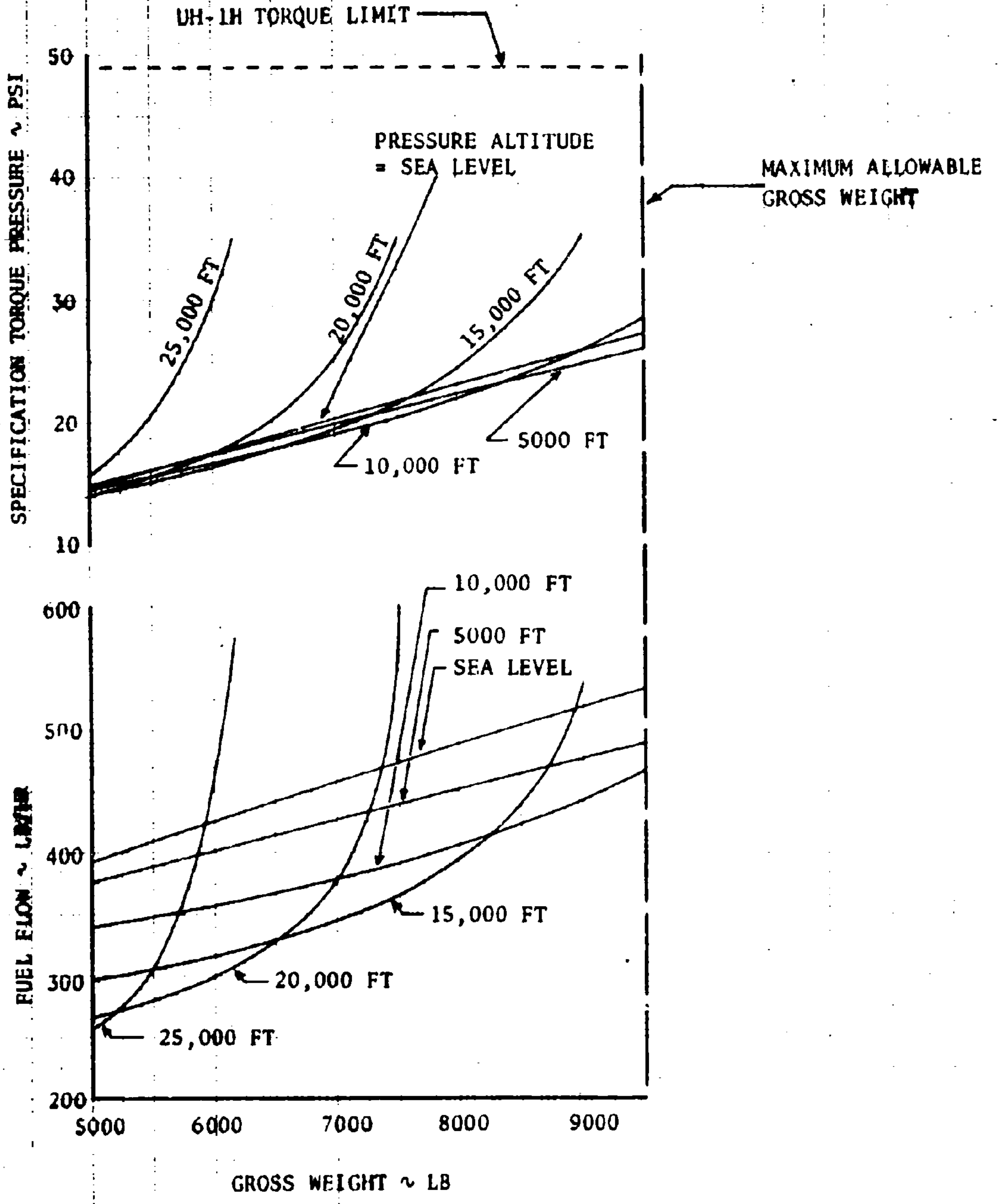


FIGURE 64
EXTRAPOLATED ENDURANCE SUMMARY
UH-1H T53-L-13

AIRSPEED = 50 KCAS ROTOR SPEED = 324 RPM
AMBIENT TEMPERATURE = -30°C
BLEED AIR HEAT FULL ON
ANTI ICE ON OR OFF

NOTE: Derived from Figures 66 and 67, and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 120

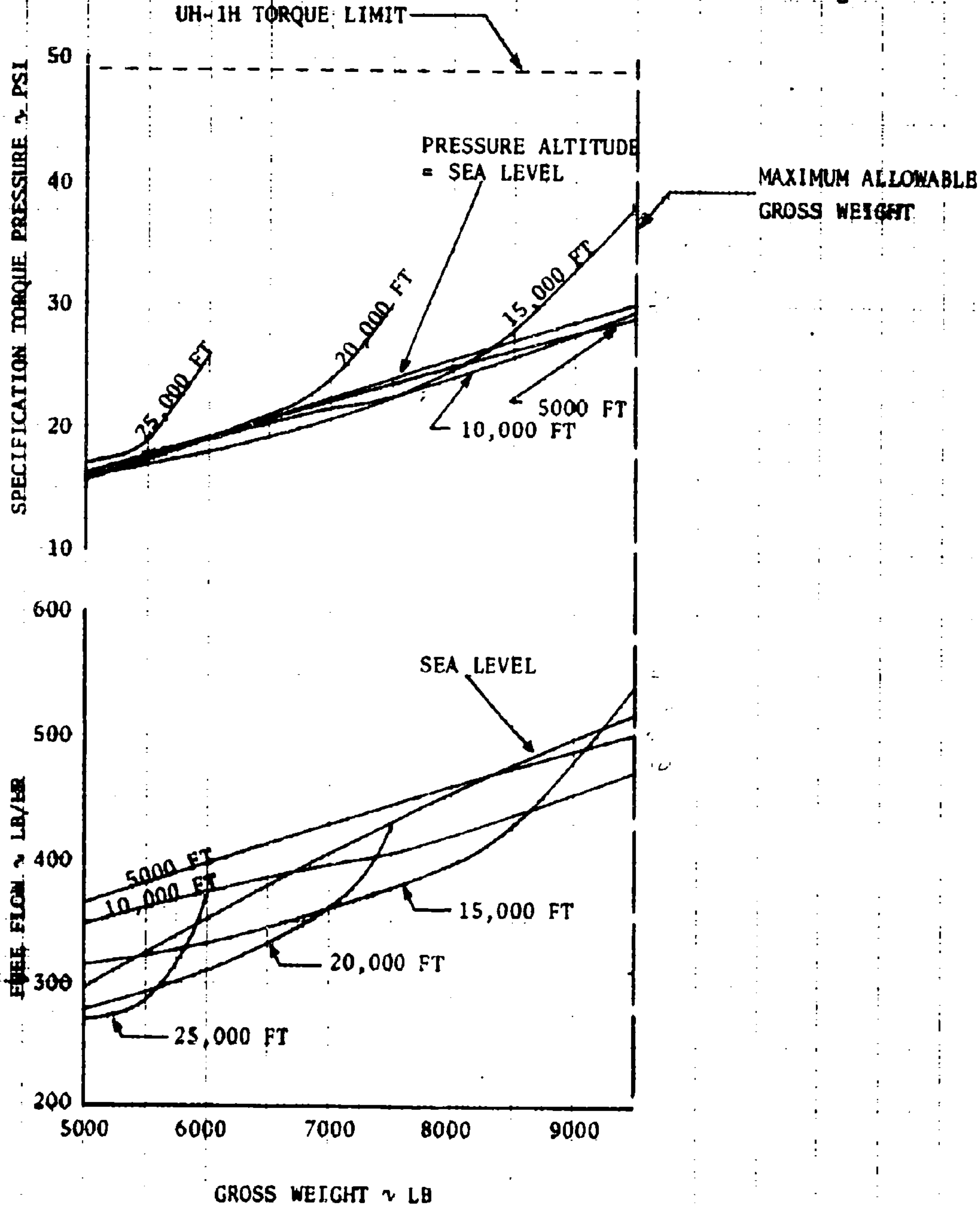


FIGURE 65
EXTRAPOLATED ENDURANCE SUMMARY
UH-1H T53-L-13

AIRSPEED = 50 KCAS ROTOR SPEED = 324 RPM
AMBIENT TEMPERATURE = -50°C
BLEED AIR HEAT FULL ON
ANTI ICE ON OR OFF

NOTE: Derived from Figures 66 and 67, and Specification Fuel Flow Corrected For Installation Losses Listed on Figure 120

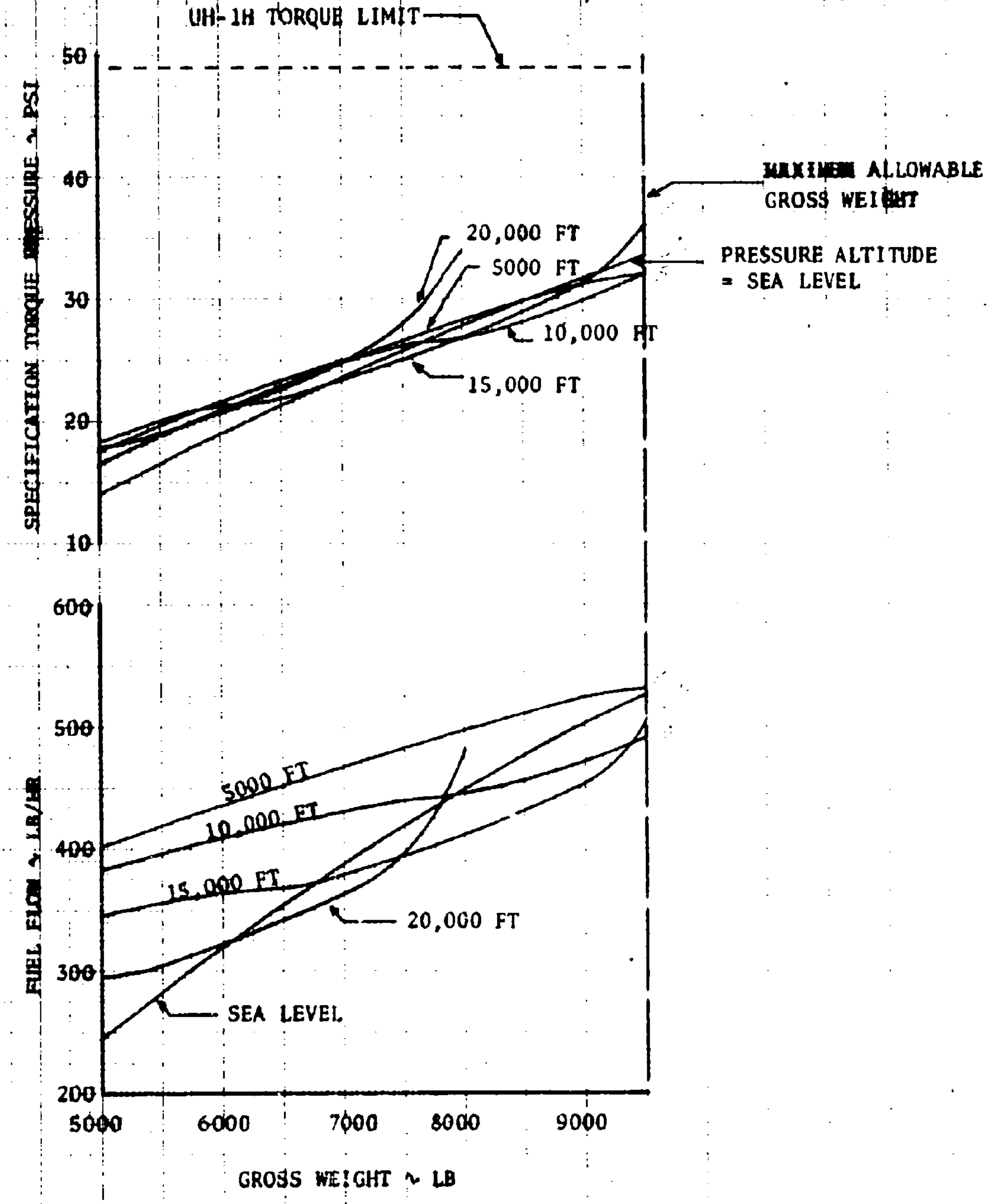
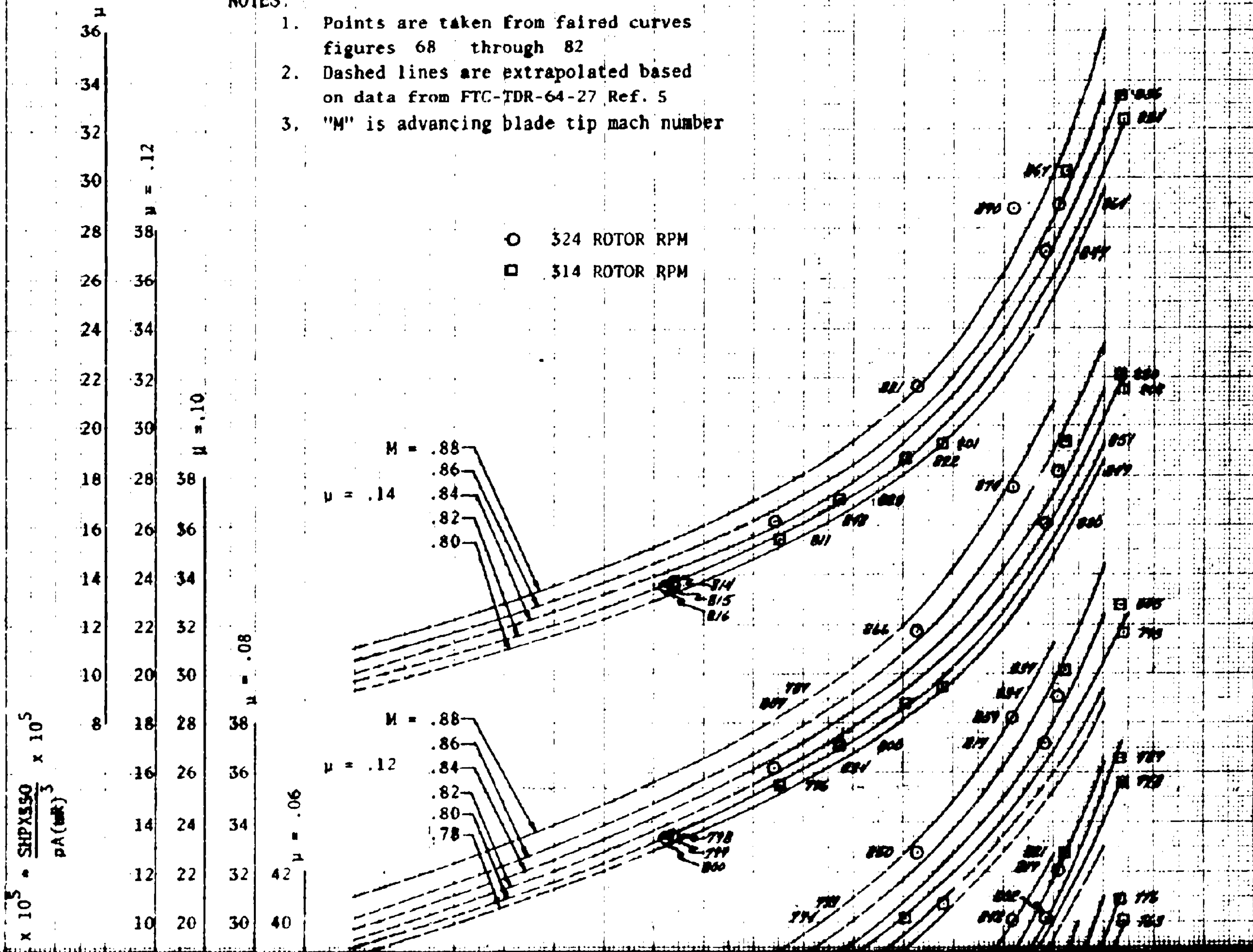


FIGURE 66
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029

NOTES:
1. Points are taken from faired curves figures 68 through 82
2. Dashed lines are extrapolated based on data from FTC-TDR-64-27 Ref. 5
3. "M" is advancing blade tip mach number



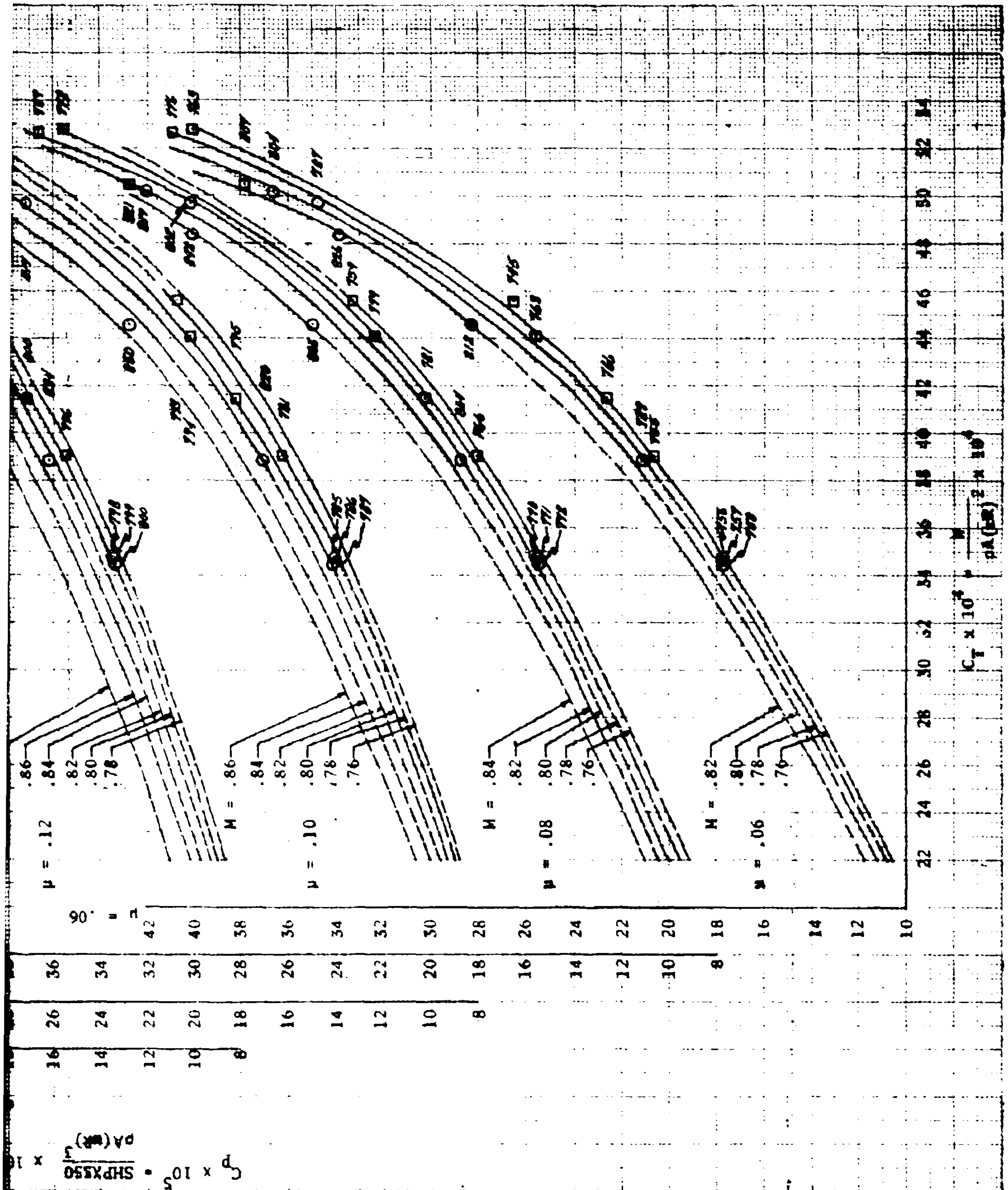
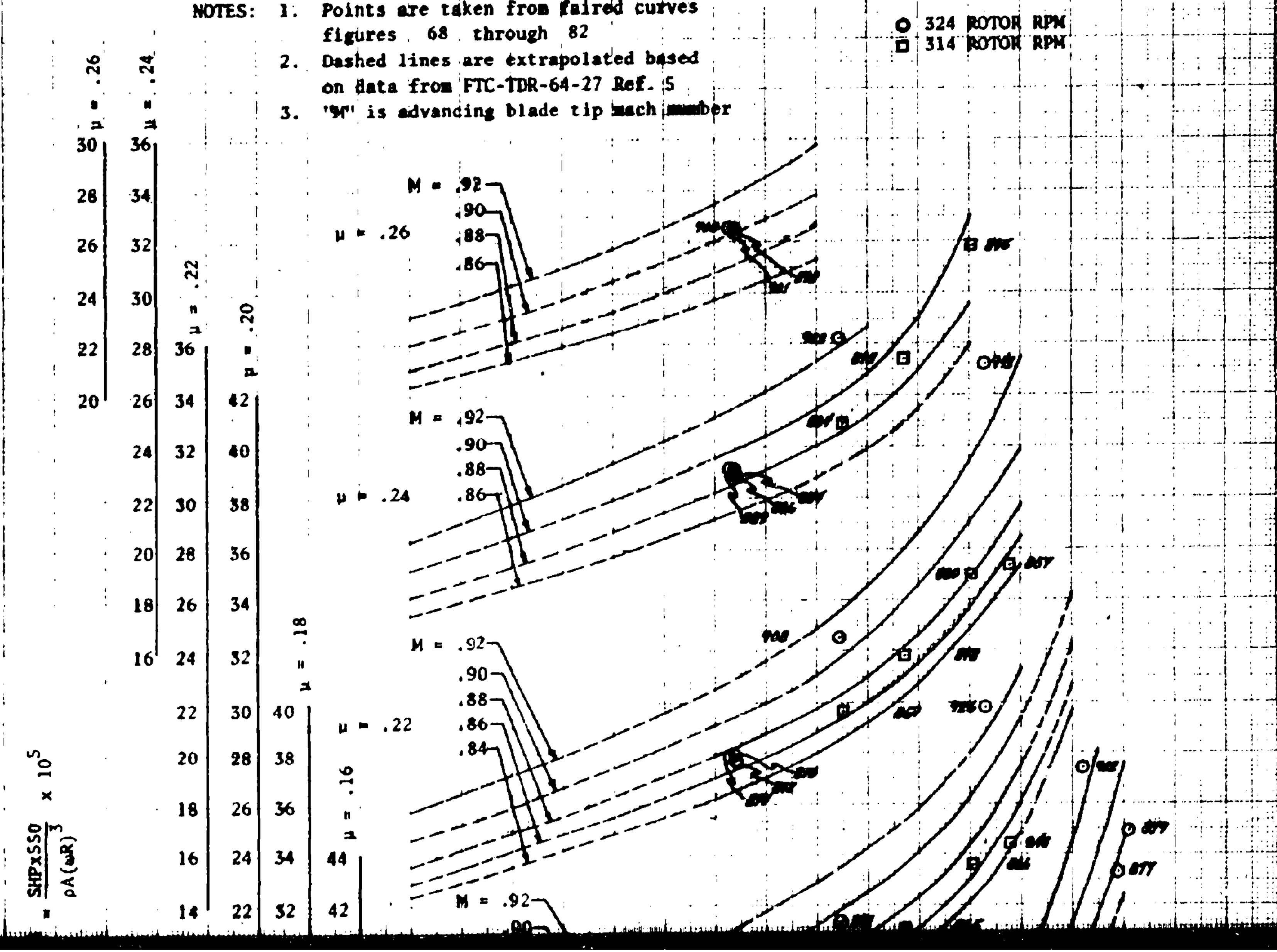
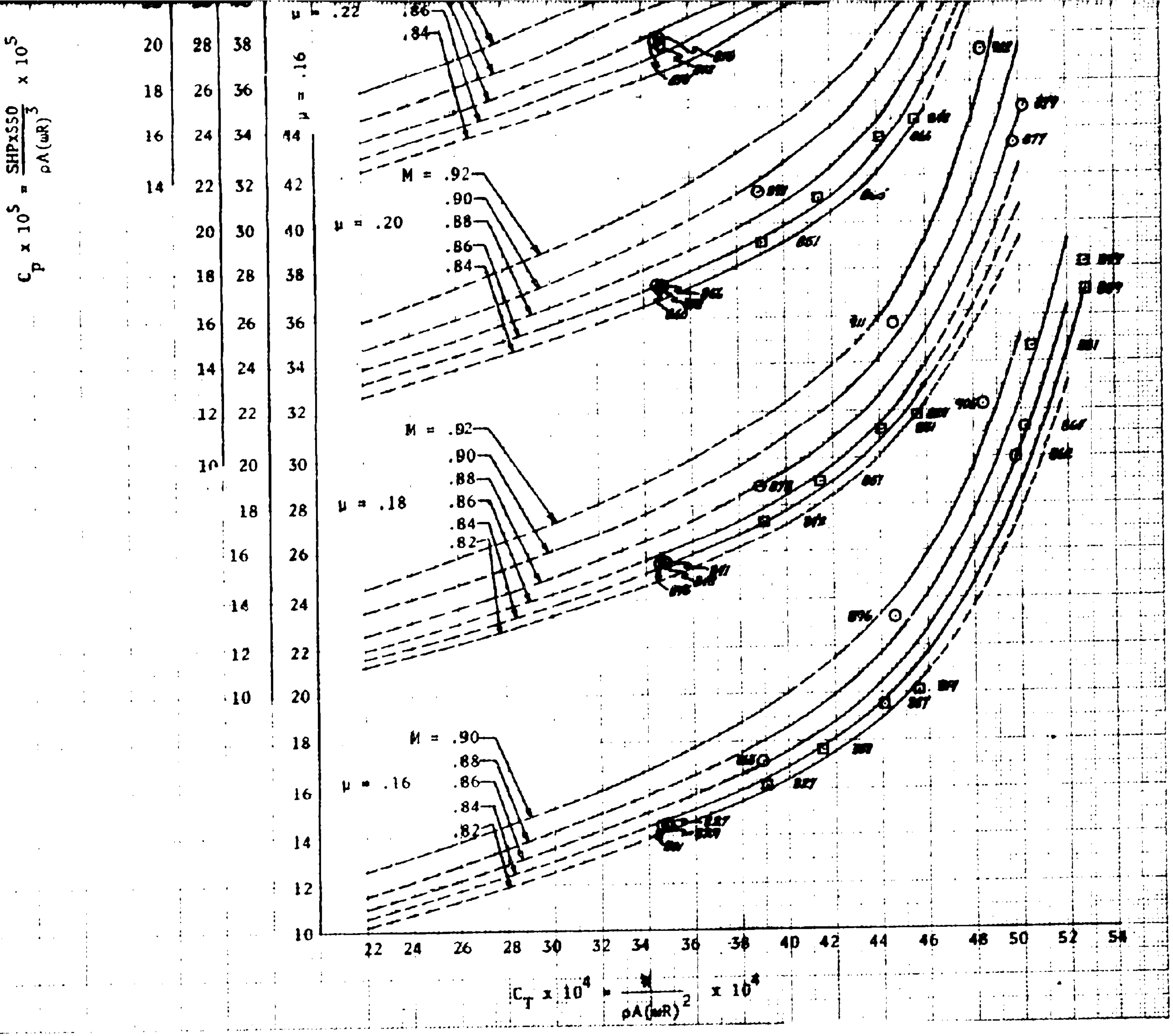

μ = .12
M = .86
.84
.82
.80
.78
μ = .10
M = .86
.84
.82
.80
.78
.76
μ = .08
M = .84
.82
.80
.78
.76
μ = .06
M = .82
.80
.78
.76
μ = .06
$C_T \times 10^4 = \frac{W}{\rho A(\Omega R)^2} \times 10^4$
$C_p \times 10^5 = \frac{SHPx550}{\rho A(\Omega R)^3} \times 10^5$

FIGURE 67
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029

NOTES:
1. Points are taken from faired curves figures 68 through 82
2. Dashed lines are extrapolated based on data from FTC-TDR-64-27 Ref. 5
3. "M" is advancing blade tip mach number





$C_p \times 10^5 = \frac{SHP \times 550}{\rho A(\omega R)^3} \times 10^5$
$C_T \times 10^4 = \frac{T}{\rho A(\omega R)^2} \times 10^4$
μ = .22
μ = .20
μ = .18
μ = .16
M = .92
M = .90

2

FIGURE 68
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14001
CLEAN CONFIGURATION ZERO SIDESLIP
AVERAGE FLIGHT CONDITIONS

C_T = 38.89 x 10^{-4} ROTOR SPEED = 324 RPM
AMBIENT TEMP = 3°C PRESSURE ALT = 5,370 FT
WEIGHT = 9,500 LB LONG C.G. = 136.3 (MID)

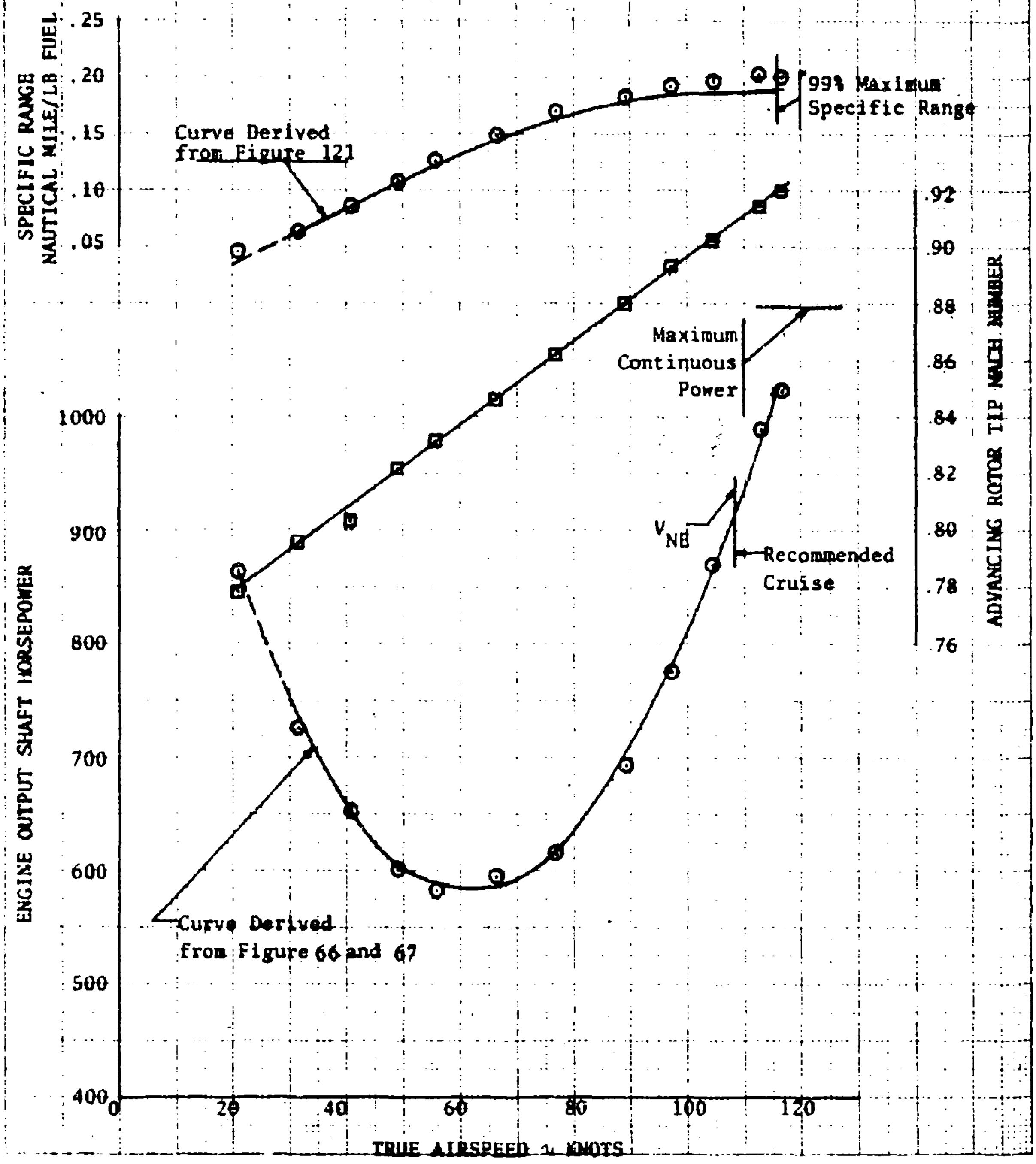


FIGURE 69

LEVEL FLIGHT PERFORMANCE

YUH-1H S/N 60-6029

T53-L-13 S/N LE-14008R

CLEAN CONFIGURATION ZERO SIDESLIP

AVERAGE FLIGHT CONDITIONS

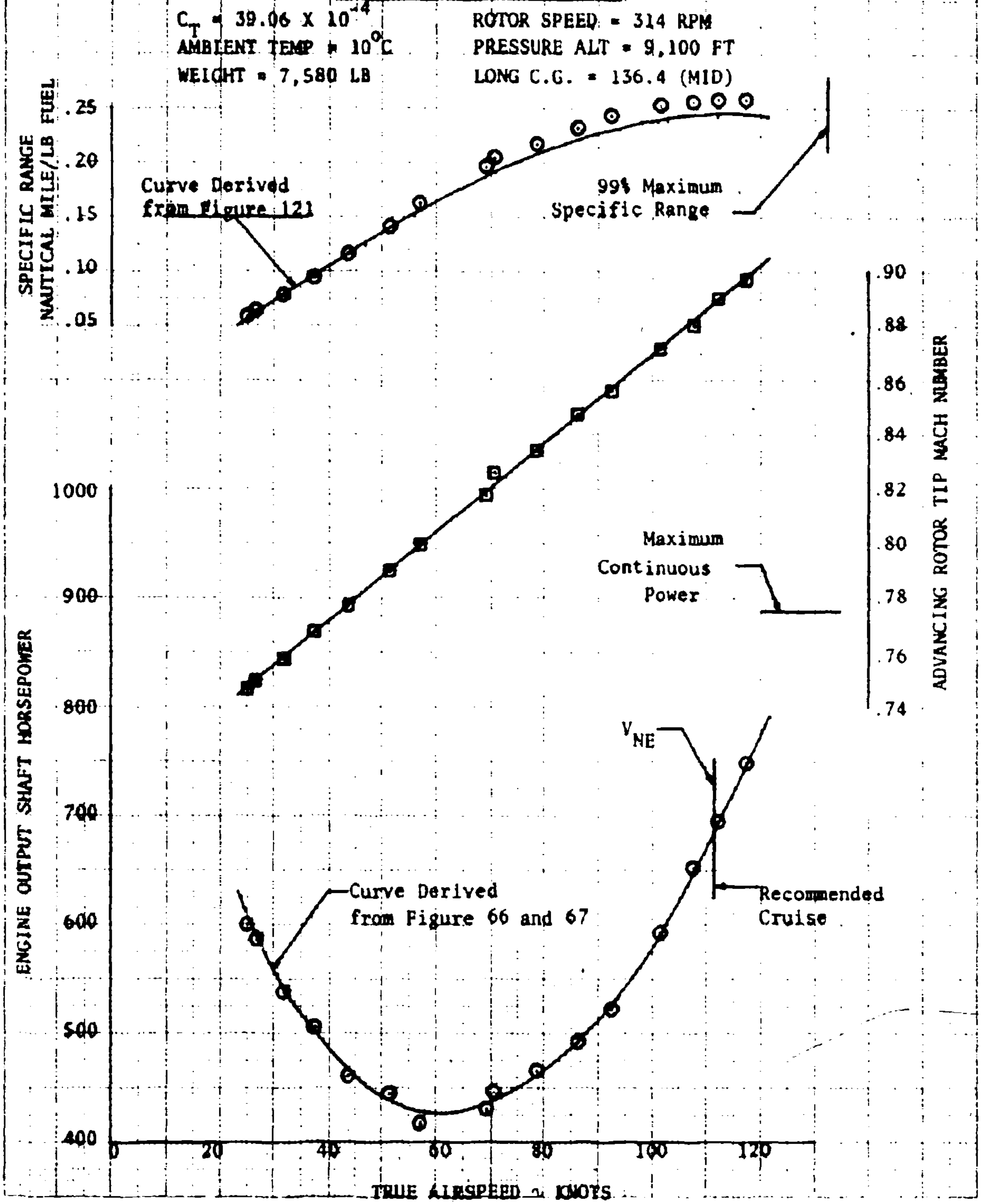


FIGURE 70
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14001
CLEAN CONFIGURATION ZERO SIDESLIP
AVERAGE FLIGHT CONDITIONS

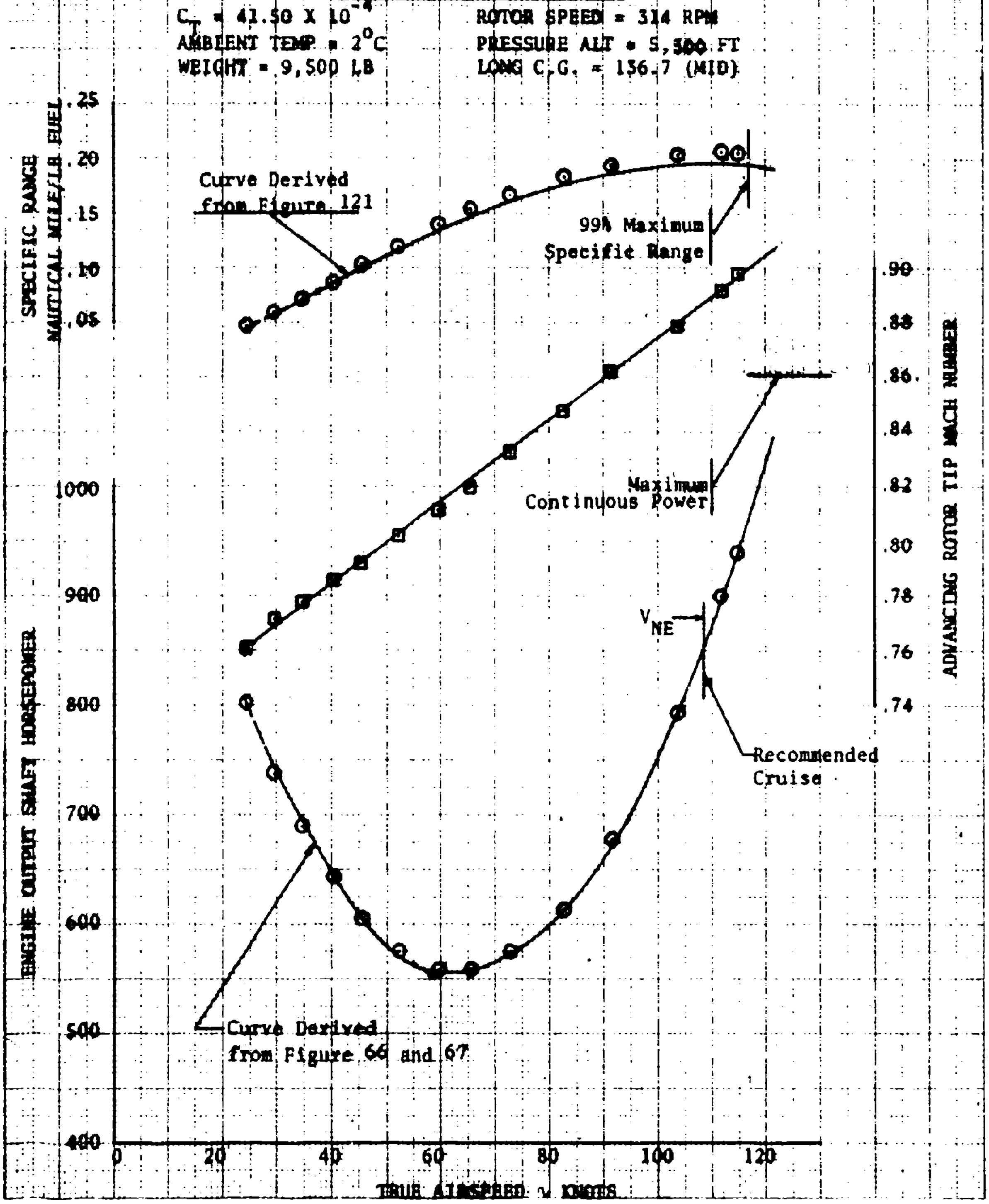


FIGURE 71

LEVEL FLIGHT PERFORMANCE

YUH-1H S/N 60-6029

T53-L-13 S/N LE-14008R

CLEAN CONFIGURATION ZERO SIDESLIP

ANTI-ICE AND BLEED AIR HEAT ON

AVERAGE FLIGHT CONDITIONS

C_T = 44.10 X 10^{-4} ROTOR SPEED = 314 RPM

AMBIENT TEMP = 3°C PRESSURE ALT = 12,120 FT

WEIGHT = 7,802 LB LONG C.G. = 137.0 (MID)

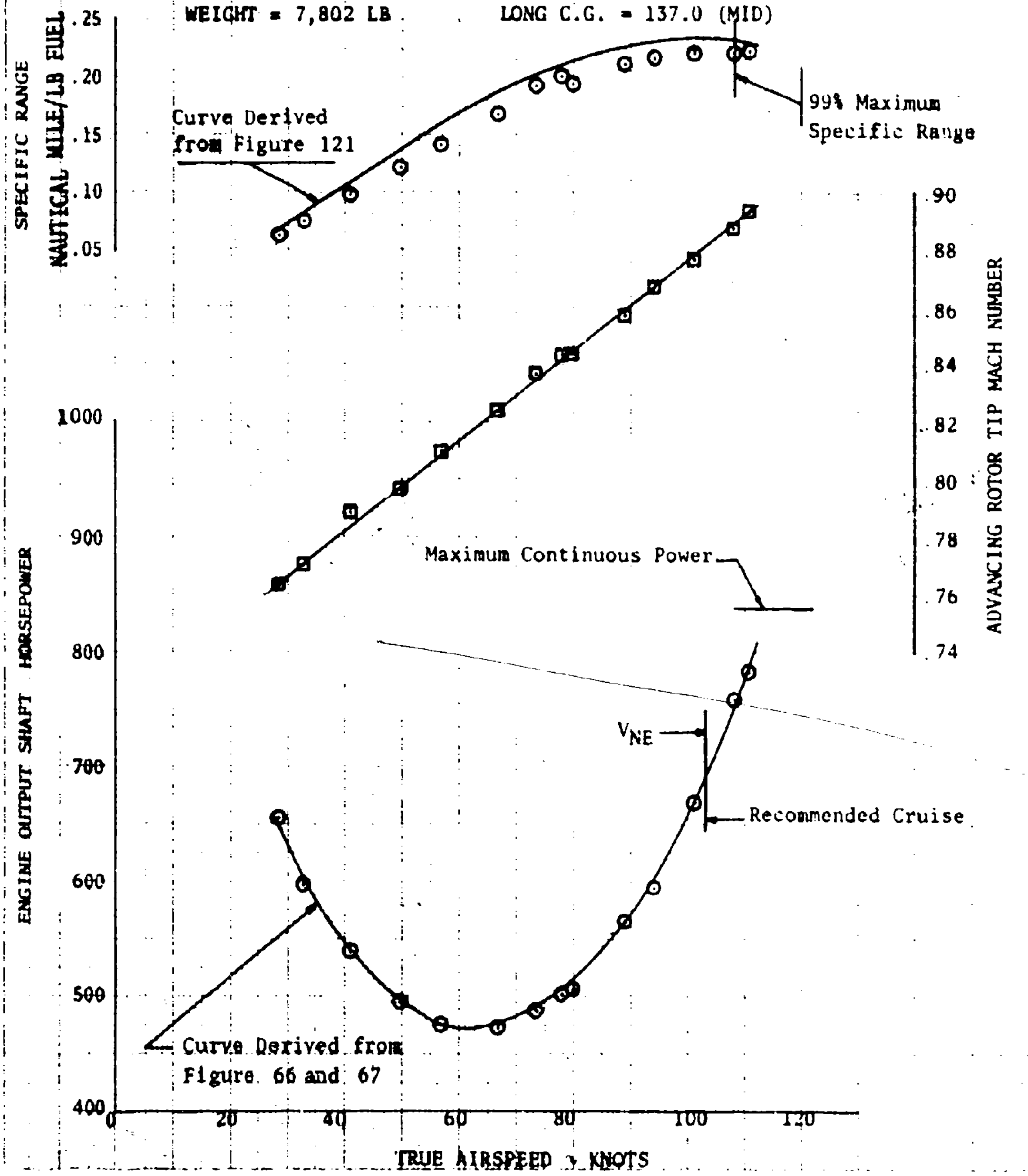
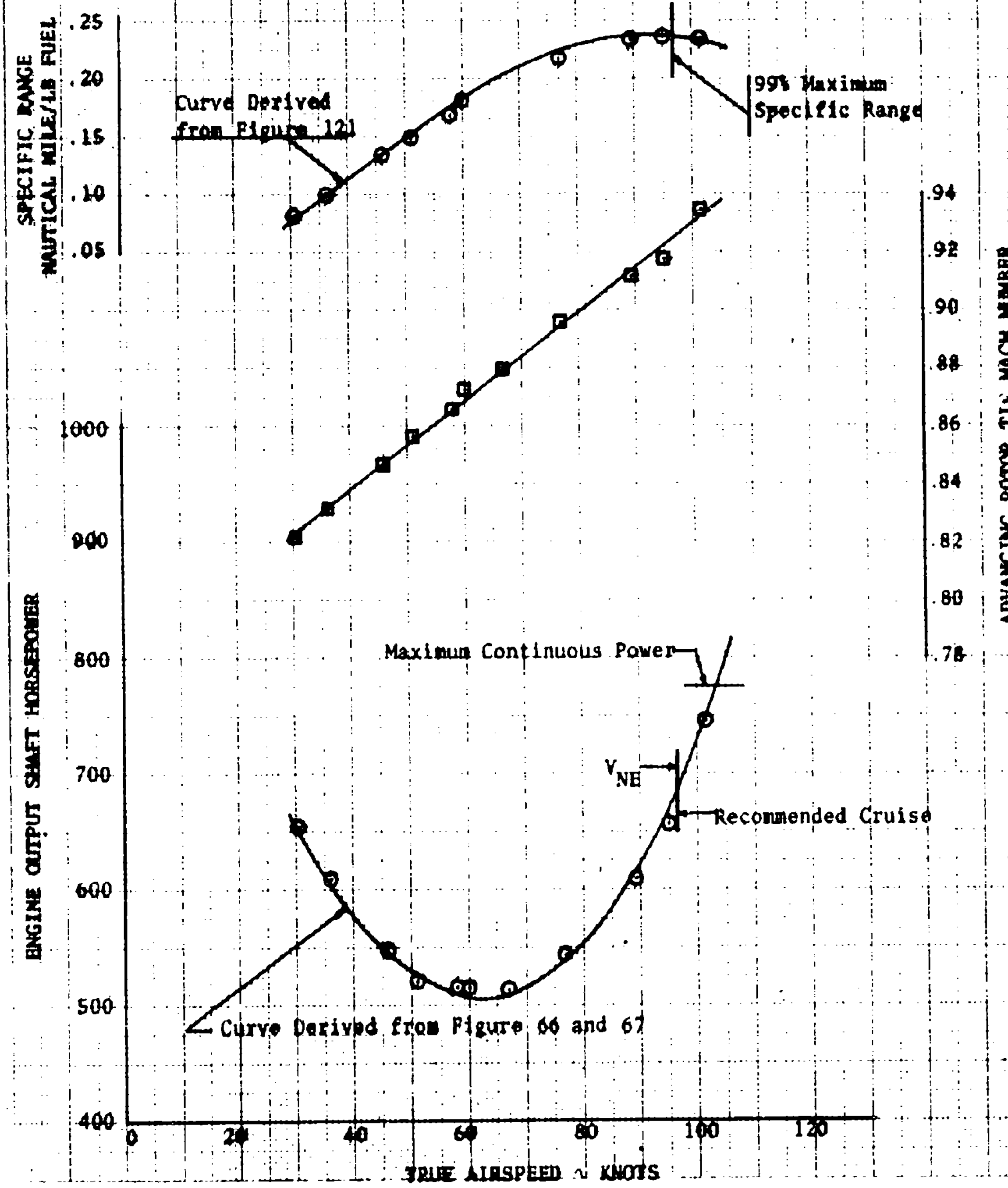


FIGURE 72
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14001
CLEAN CONFIGURATION ZERO SIDESLIP
AVERAGE FLIGHT CONDITIONS

C_T = 44.61 X 10 ROTOR SPEED = 324 RPM
AMBIENT TEMP = -17°C PRESSURE ALT = 18,230 FT
WEIGHT = 7,0[illegible]0 LB LONG C.G. = 136.8 (MID)



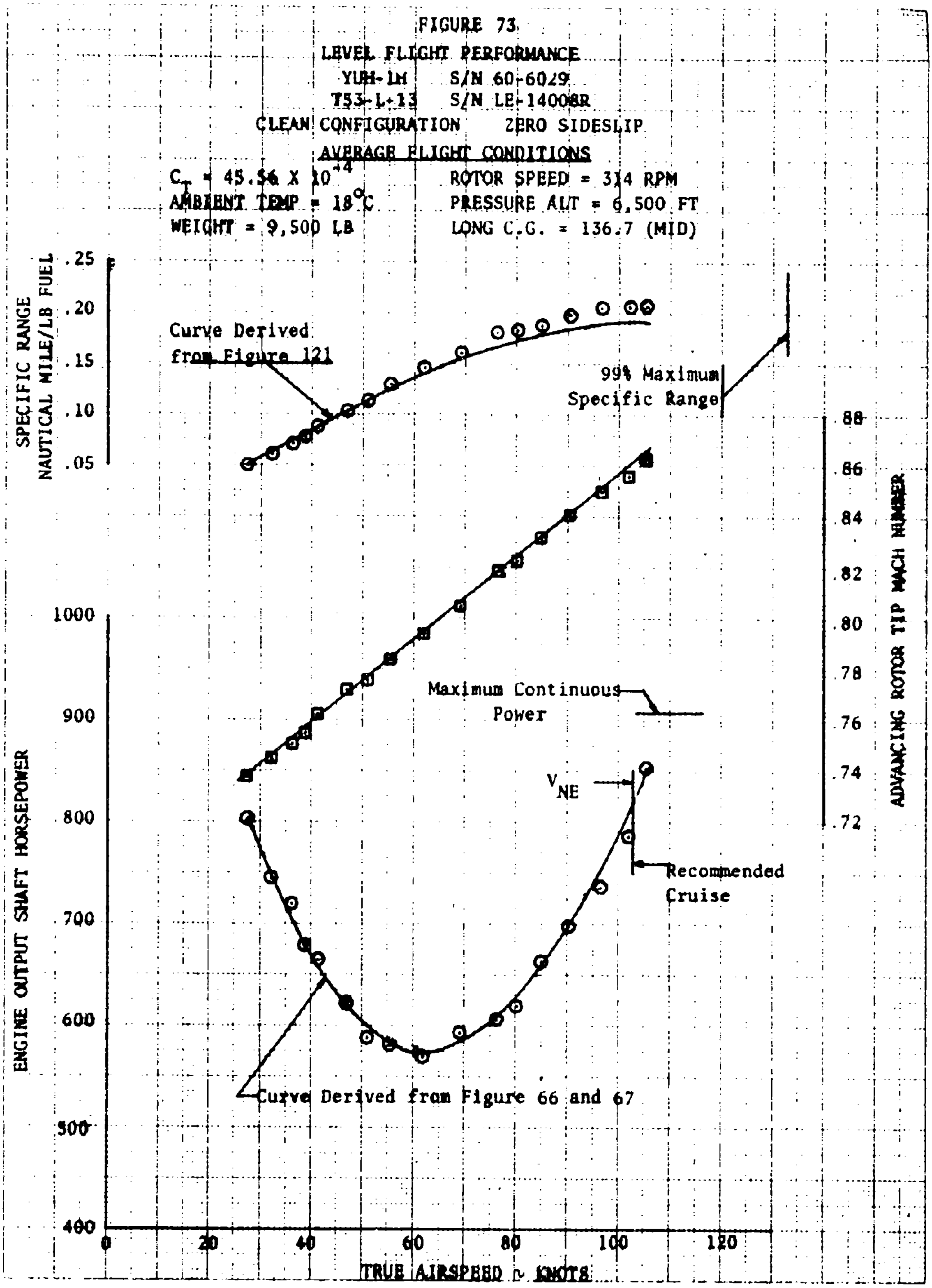


FIGURE 73
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008R
CLEAN CONFIGURATION ZERO SIDESLIP
AVERAGE FLIGHT CONDITIONS
C_T = 45.56 X 10^-4
ROTOR SPEED = 314 RPM
AMBIENT TEMP = 18°C
PRESSURE ALT = 6,500 FT
WEIGHT = 9,500 LB
LONG C.G. = 136.7 (MID)
SPECIFIC RANGE
NAUTICAL MILE/LB FUEL
.25
.20
.15
.10
.05
Curve Derived from Figure 121
99% Maximum Specific Range
ADVANCING ROTOR TIP MACH NUMBER
.88
.86
.84
.82
.80
.78
.76
.74
.72
Maximum Continuous Power
V_NE
Recommended Cruise
Curve Derived from Figure 66 and 67
ENGINE OUTPUT SHAFT HORSEPOWER
1000
900
800
700
600
500
400
0
20
40
60
80
100
120
TRUE AIRSPEED ~ KNOTS

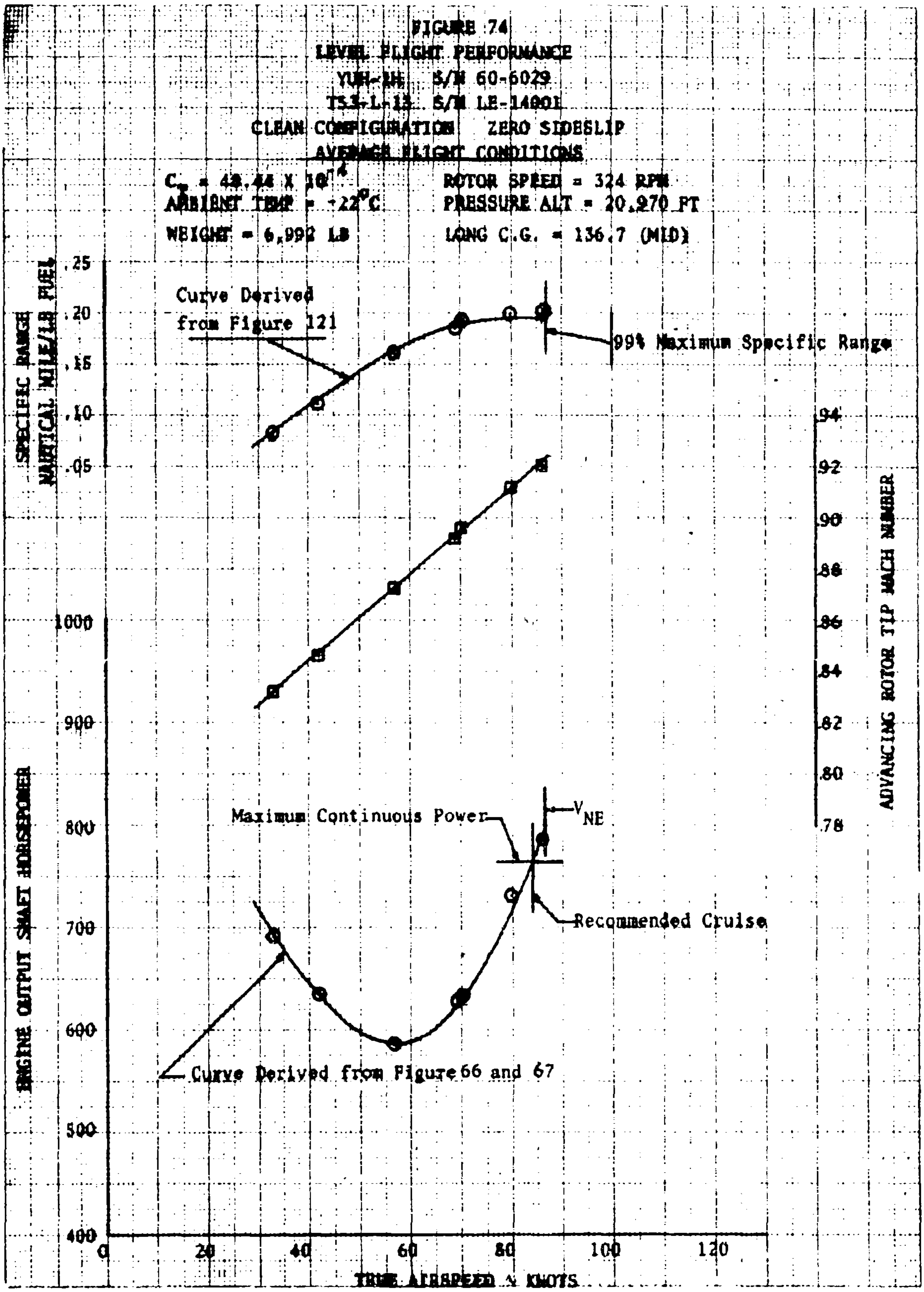


FIGURE 74
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14001
CLEAN CONFIGURATION ZERO SIDESLIP
AVERAGE FLIGHT CONDITIONS
ROTOR SPEED = 324 RPM
AMBIENT TEMP = -22°C
PRESSURE ALT = 20,970 FT
WEIGHT = 6,992 LB
LONG C.G. = 136.7 (MID)
Curve Derived from Figure 121
99% Maximum Specific Range
SPECIFIC RANGE NAUTICAL MILE/LB FUEL
ADVANCING ROTOR TIP MACH NUMBER
Maximum Continuous Power
V_{NE}
Recommended Cruise
Curve Derived from Figure 66 and 67
ENGINE OUTPUT SHAFT HORSEPOWER
TRUE AIRSPEED ~ KNOTS

FIGURE 75
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14001
CLEAN CONFIGURATION ZERO SIDESLIP

AVERAGE FLIGHT CONDITIONS

$C_T = 49.74 \times 10^{-4}$ ROTOR SPEED = 324 RPM
AMBIENT TEMP = 4°C PRESSURE ALT = 11,680 FT
WEIGHT 9,500 LB LONG C.G. = 136.1 (MID)

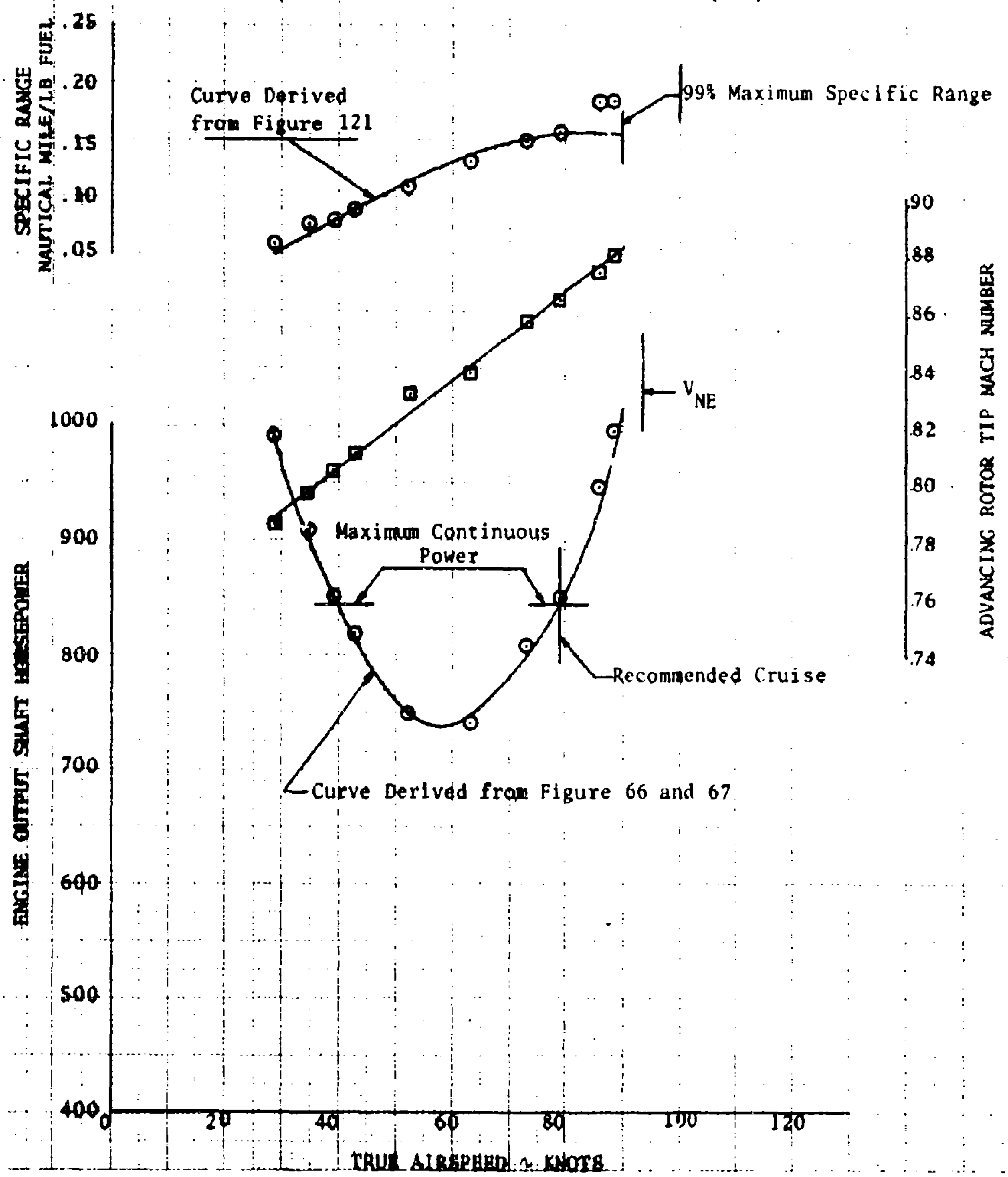


FIGURE 76
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008
CLEAN CONFIGURATION ZERO SIDESLIP

AVERAGE FLIGHT CONDITIONS

C_T = 50.21 X 10^{-4}
AMBIENT TEMP = -7°C
WEIGHT = 8,[illegible] LB
ROTOR SPEED = 324 RPM
PRESSURE ALT = 15,080 FT
LONG C.G. = 136.6 (MID)

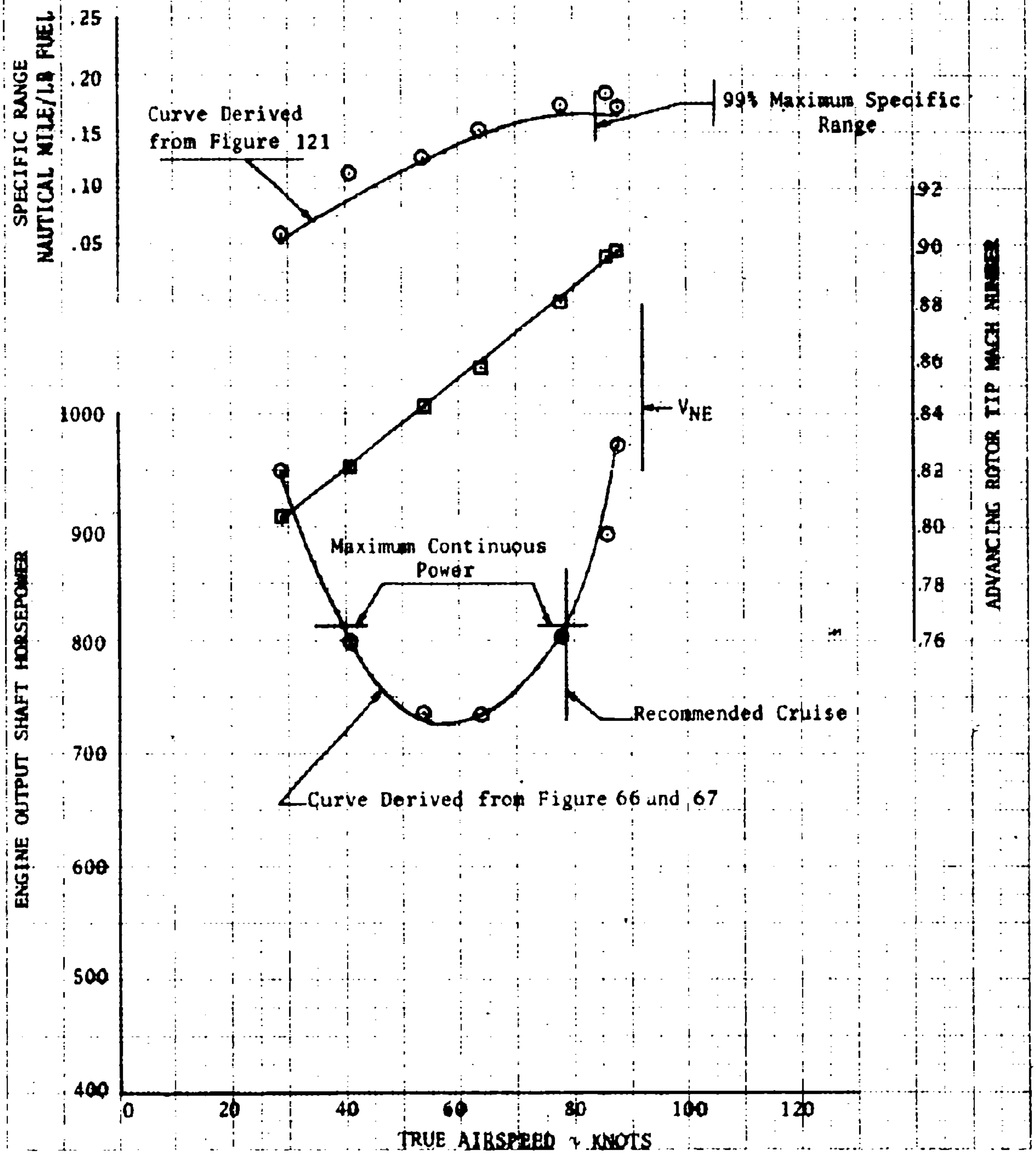


FIGURE 77
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14001
CLEAN CONFIGURATION ZERO SIDESLIP

AVERAGE FLIGHT CONDITIONS

C_T = 50.94 X 10^{-4} ROTOR SPEED = 314 RPM
AMBIENT TEMP = -25°C PRESSURE ALT = 20,790 FT
WEIGHT = 6,890 LB LONG C.G. = 136.9 (MID)

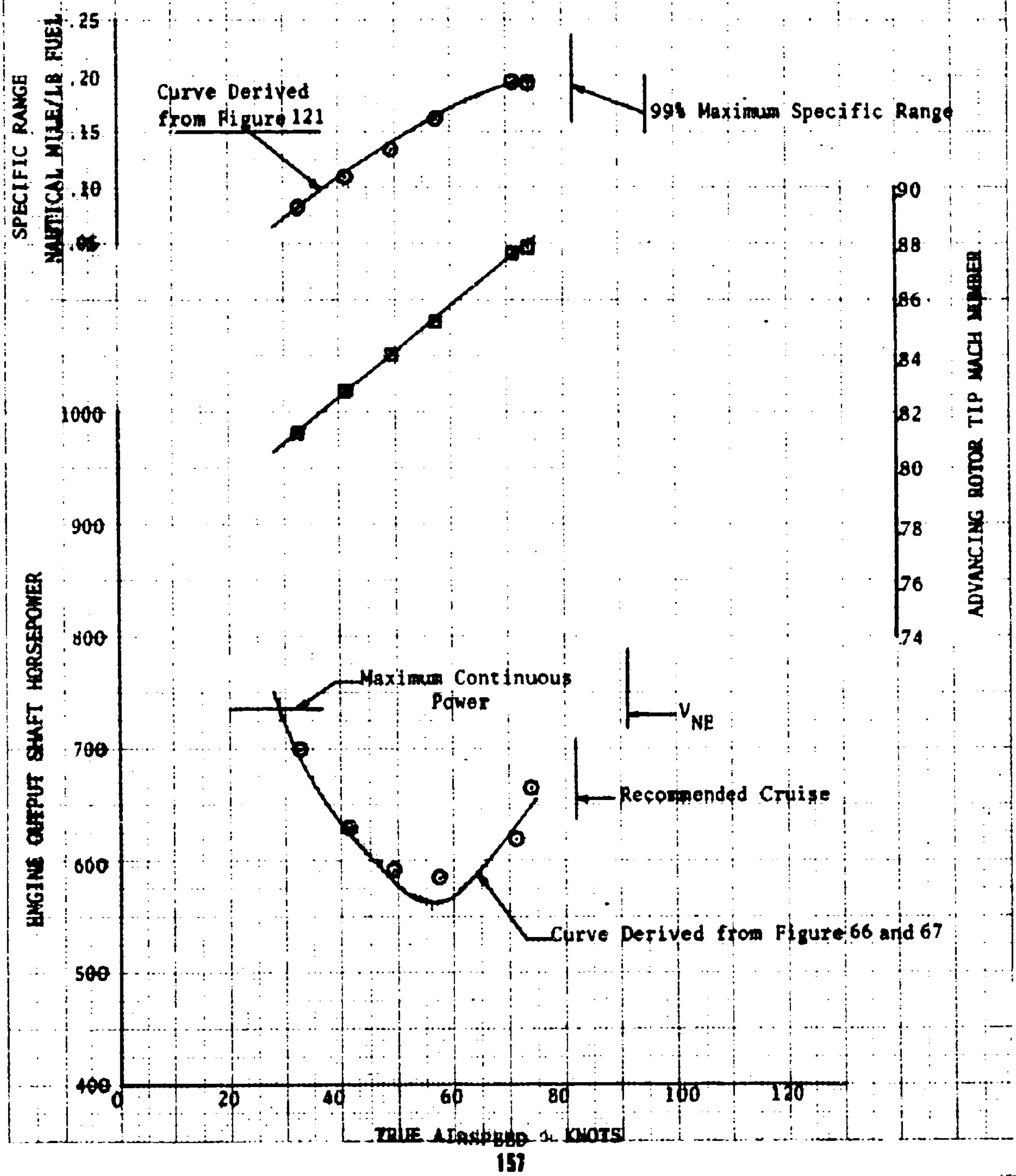


FIGURE 78
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14001
CLEAN CONFIGURATION ZERO SIDESLIP
AVERAGE FLIGHT CONDITIONS

C_T = 52.75 X 10^{-4} ROTOR SPEED = 314 RPM
AMBIENT TEMP = -5.5°C PRESSURE ALT = 14,950 FT
WEIGHT 8,600 LB LONG C.G. = 136.9 (MID)

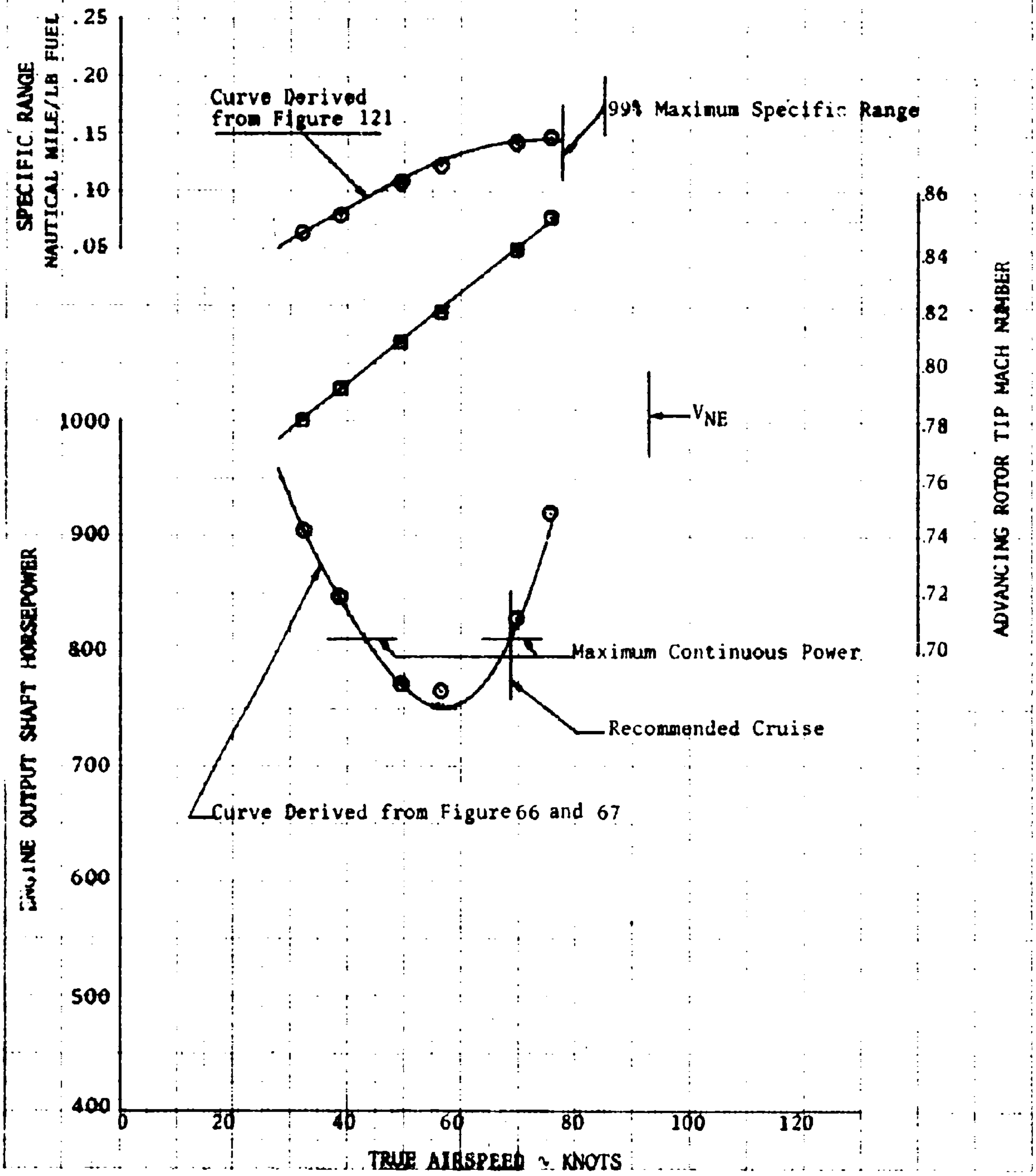


FIGURE 79
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14001
CLEAN CONFIGURATION ZERO SIDESLIP
AVERAGE FLIGHT CONDITIONS

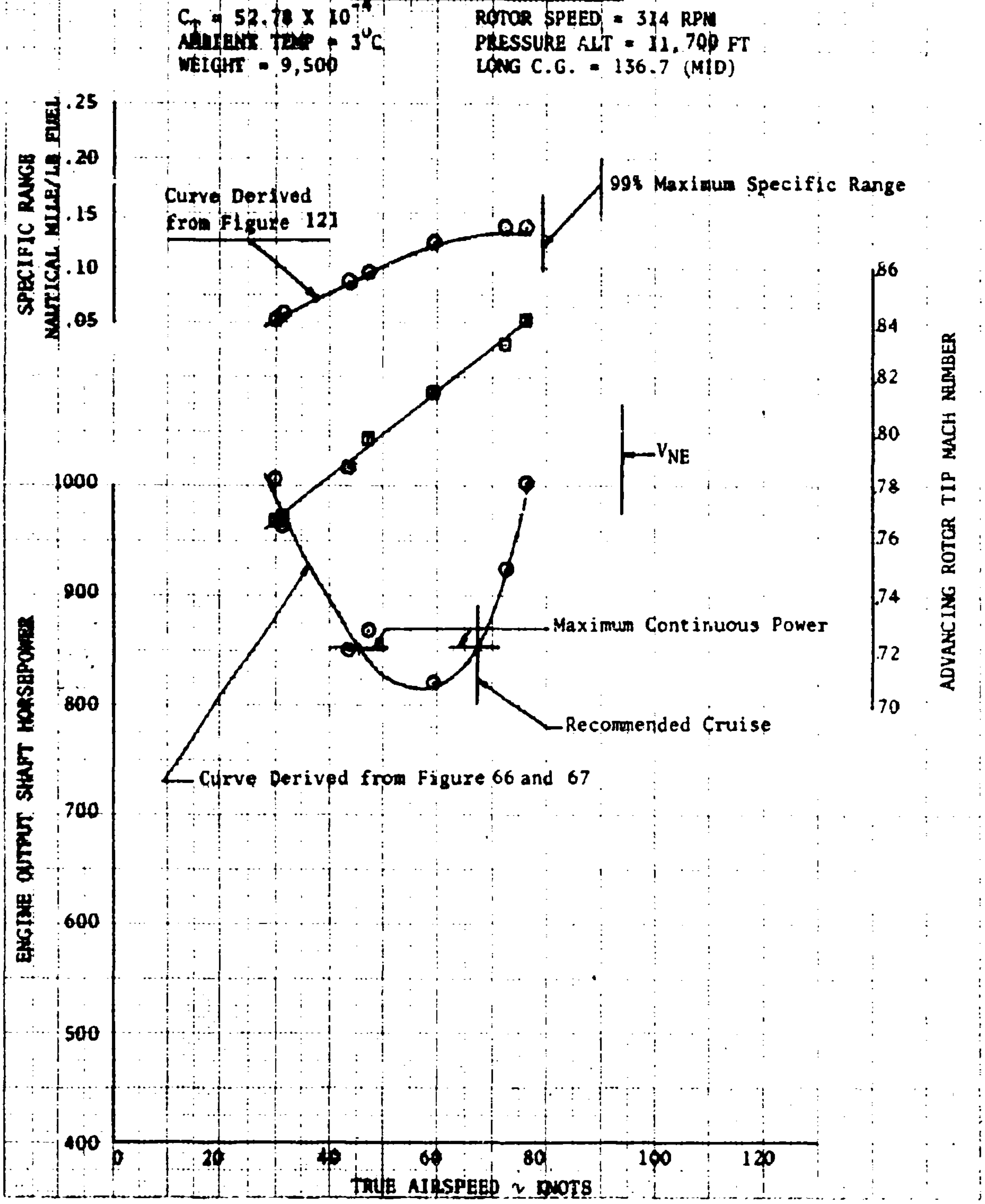
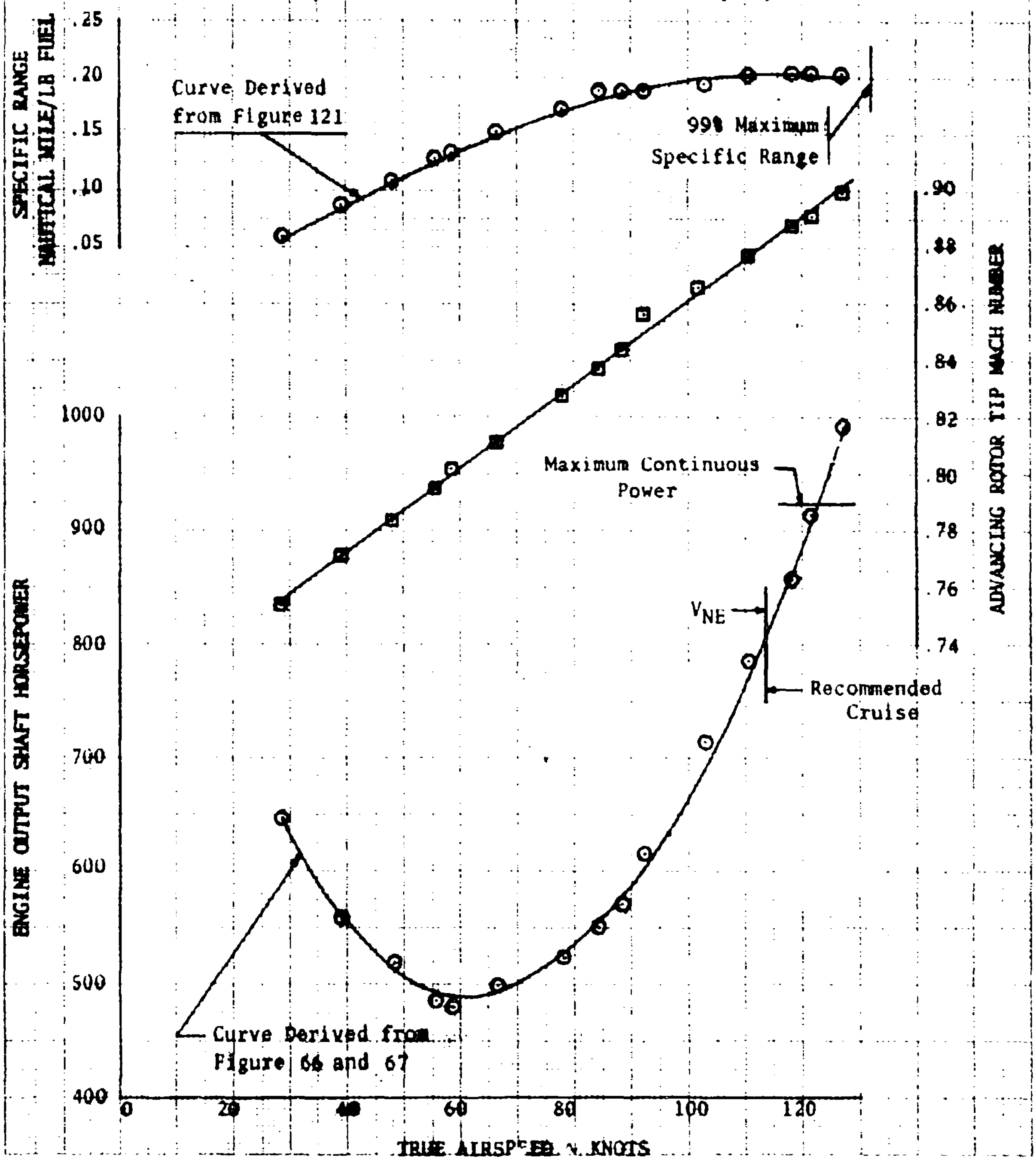


FIGURE 80
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008R
CLEAN CONFIGURATION ZERO SIDESLIP

AVERAGE FLIGHT CONDITIONS

$C_T = 34.85 \times 10^{-4}$ ROTOR SPEED = 324 RPM
AMBIENT TEMP = 28°C PRESSURE ALT = 2,860 FT
WEIGHT = 8,560 LB LONG C.G. = 130.1 (FWD)



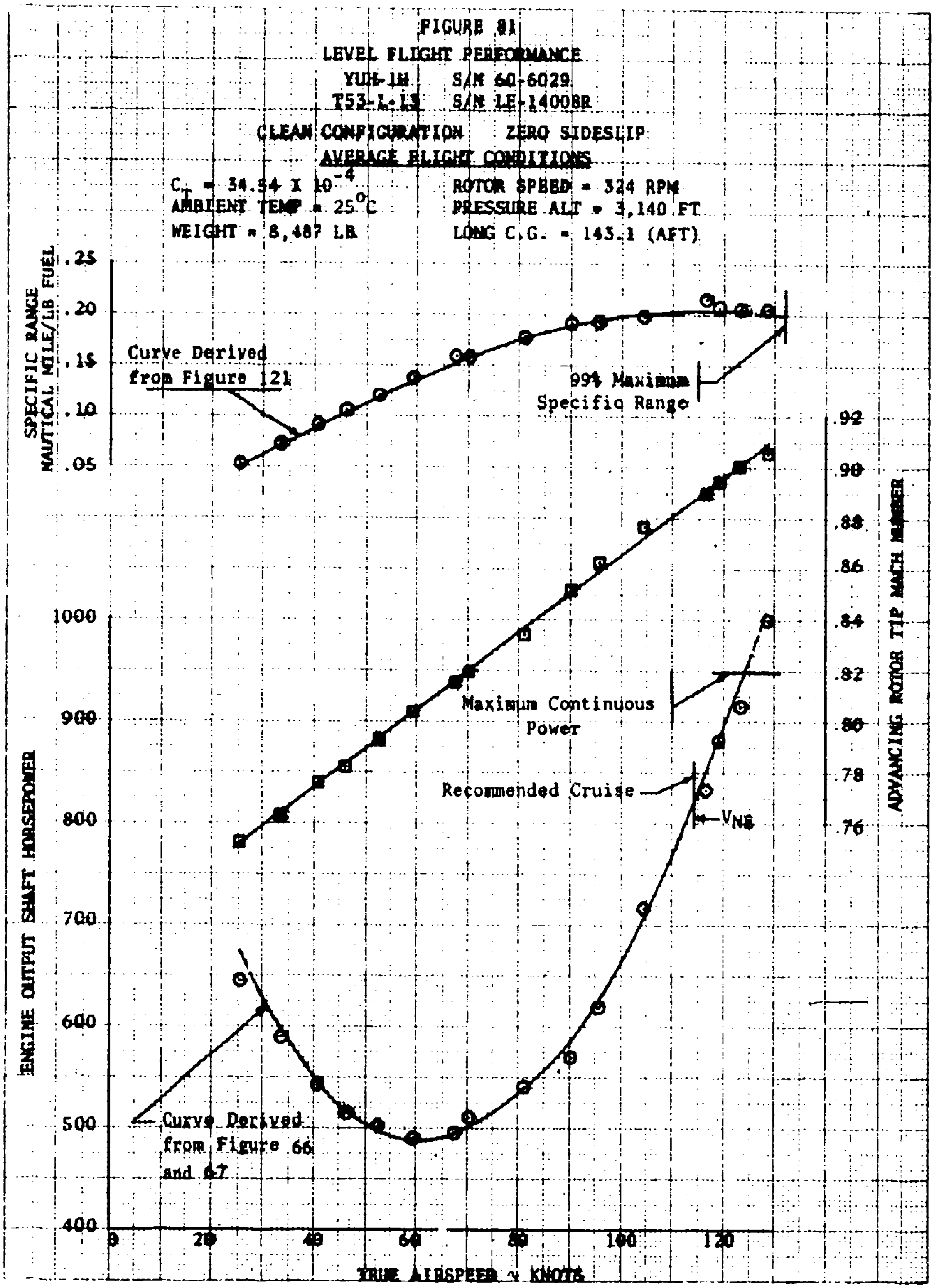


FIGURE 91
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008R
CLEAN CONFIGURATION ZERO SIDESLIP
AVERAGE FLIGHT CONDITIONS
C_T = 34.54 X 10^{-4}
AMBIENT TEMP = 25°C
WEIGHT = 8,487 LB
ROTOR SPEED = 324 RPM
PRESSURE ALT = 3,140 FT
LONG C.G. = 143.1 (AFT)
SPECIFIC RANGE
NAUTICAL MILE/LB FUEL
Curve Derived from Figure 121
99% Maximum Specific Range
Maximum Continuous Power
Recommended Cruise
V_{NE}
Curve Derived from Figure 66 and 67
ENGINE OUTPUT SHAFT HORSEPOWER
ADVANCING ROTOR TIP MACH NUMBER
TRUE AIRSPEED ~ KNOTS

FIGURE 82
LEVEL FLIGHT PERFORMANCE
YUH-1H S/N 60-6029
T53-L-13 S/N LE-14008R
CARGO DOORS OPEN ZERO SIDESLIP
AVERAGE FLIGHT CONDITIONS

C_T = 34.80 X 10^{-4} ROTOR SPEED = 324 RPM
AMBIENT TEMP = 26.5°C PRESSURE ALT = 2,980 FT
WEIGHT = 8,560 LB LONG C.G. = 137.0 (MID)

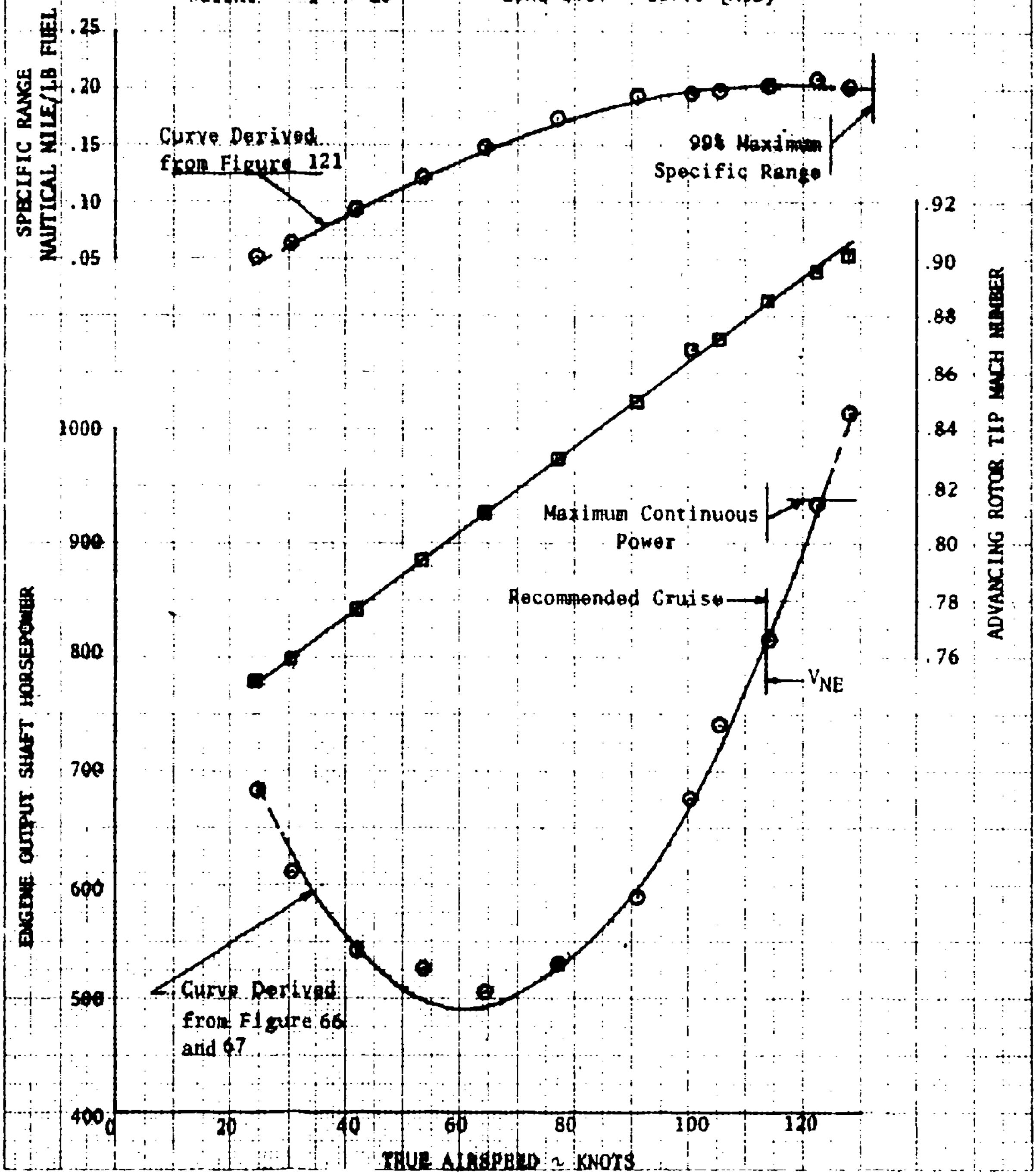


FIGURE 83
AIRSPEED CALIBRATION
YUH-1H USA S/N 60-6029
LEVEL FLIGHT

GROSS WEIGHT ~LBS	C.G. STATION ~INCHES	DENSITY ALTITUDE ~FEET	ROTOR SPEED RPM	CONFIGURATION
7050	136.4 (MID)	1130	324	CLEAN

STANDARD SYSTEM (NOSE MOUNTED PITOT PROBE)

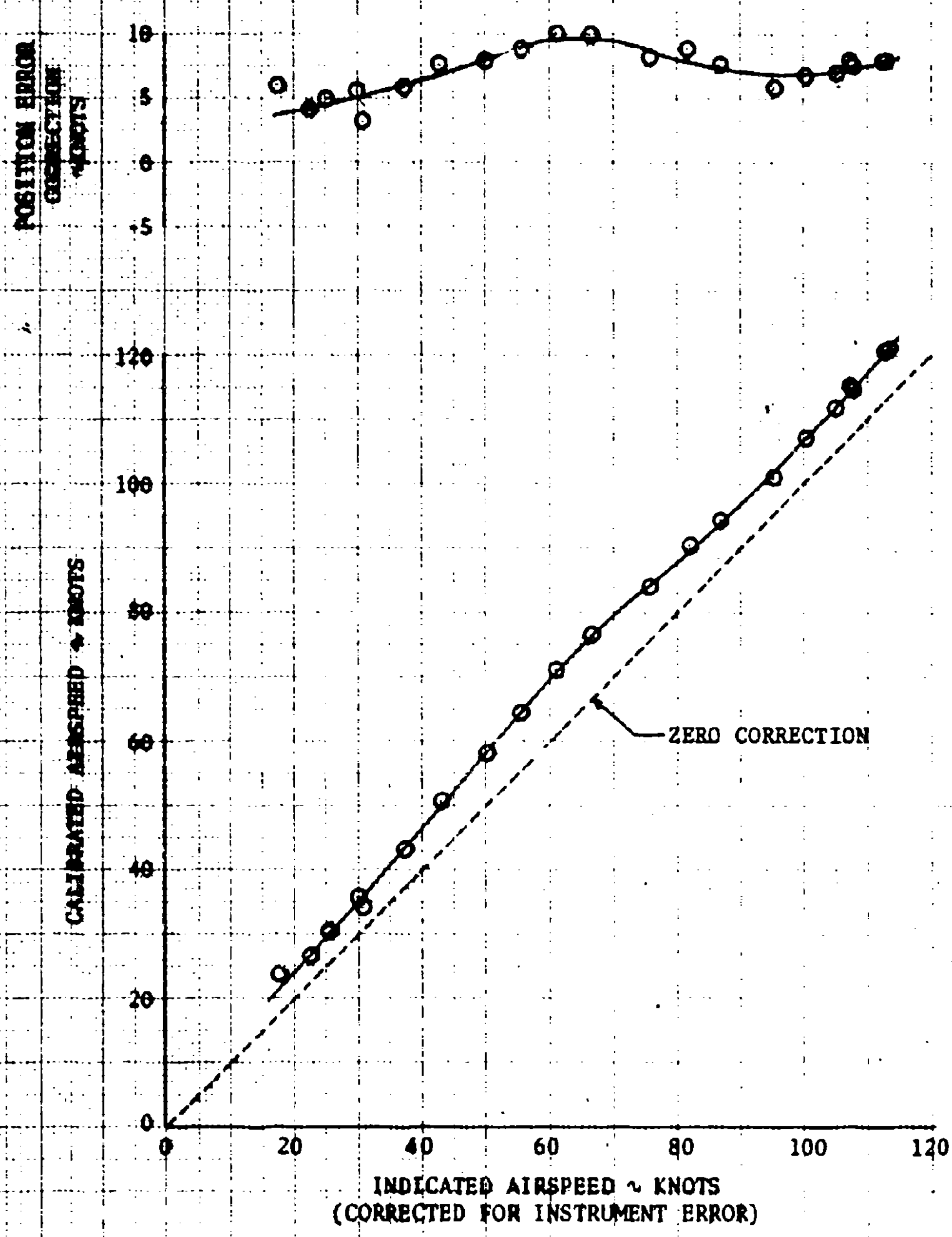


FIGURE 84
AIRSPEED CALIBRATION
YUH-1H USA S/N 60-6029
LEVEL FLIGHT

GROSS WEIGHT ~LBS	C.G. STATION ~INCHES	DENSITY ALTITUDE ~FEET	ROTOR SPEED RPM	CONFIGURATION
7050	136.4 (MID)	1130	324	CLEAN

BOOM TEST SYSTEM

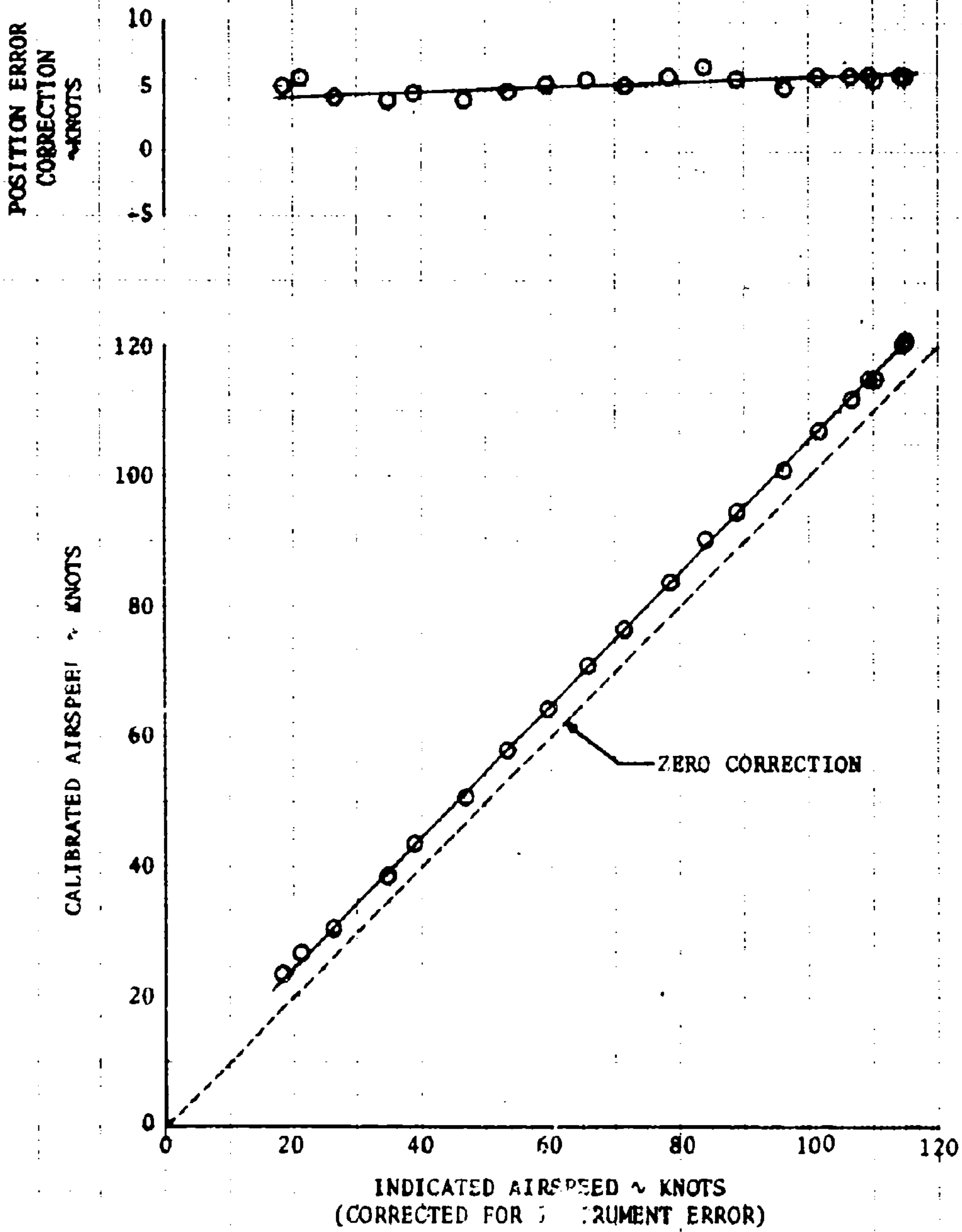


FIGURE 85
PEDAL POSITION VARIATION
IN SIDEWARD FLIGHT
YUH-1H S/N 60-6029

NOTE: Points Obtained From Figures 87 Through 101

SYMBOL	WEIGHT	LONG CG	DENSITY ALTITUDE
□	8600 LB	130.2 (FWD)	4080 FT
⦶	9500 LB	137.1 (MID)	4250 FT
△	9500 LB	134.0 (FWD)	3750 FT
⦵	8300 LB	136.2	4840 FT
▲	7000 LB	130.0 (FWD)	5400 FT
○	8600 LB	130.2 (FWD)	12,760 FT
◇	8600 LB	144.0 (AFT)	4780 FT
▽	9500 LB	143.9 (AFT)	5700 FT

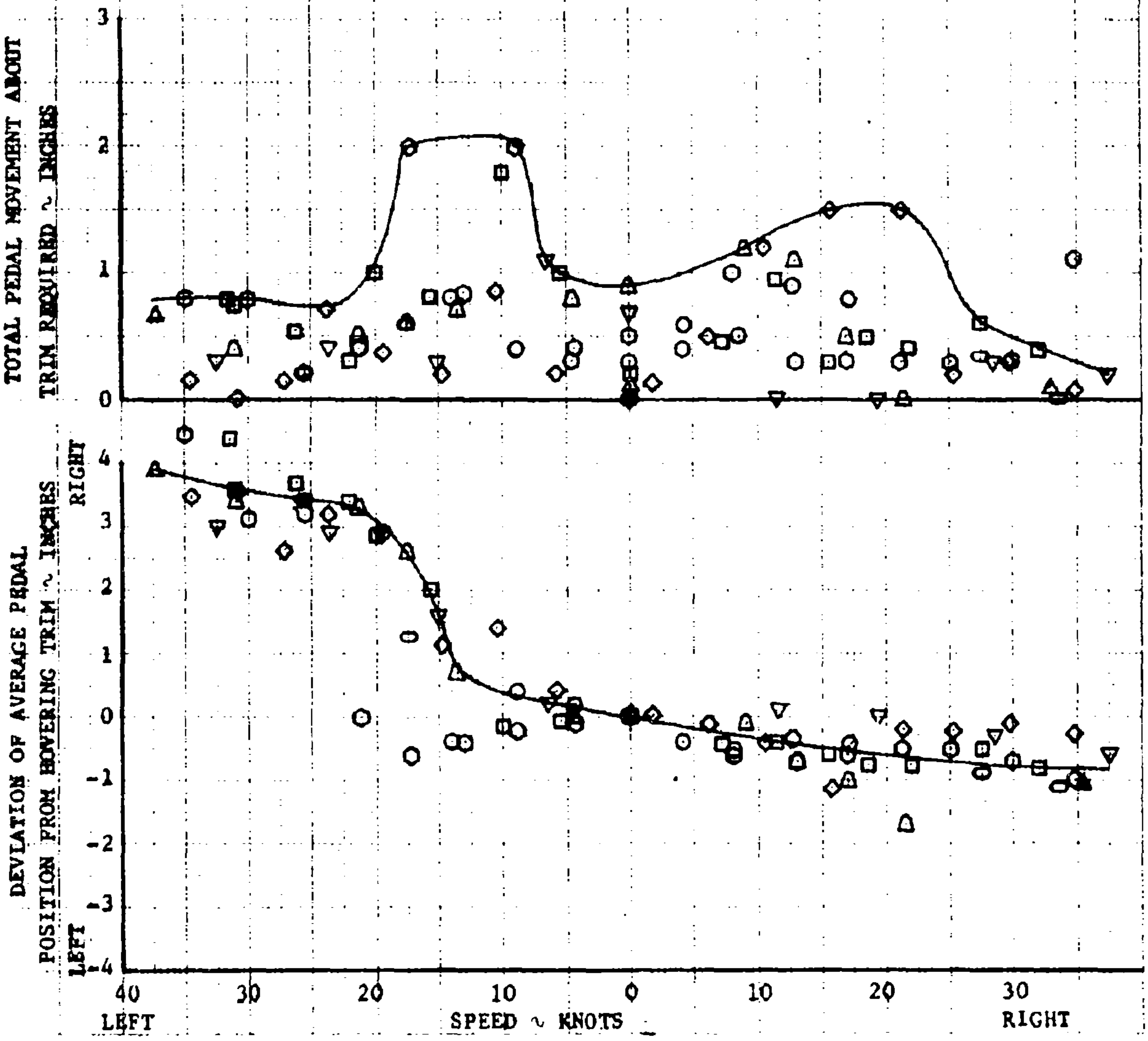


FIGURE 86
LONGITUDINAL CYCLIC POSITION VARIATION
IN REARWARD FLIGHT
YUH-1H S/N 60-6029

NOTE: Points Obtained From Figures 88 Through 102

SYMBOL	WEIGHT	LONG CG	DENSITY ALTITUDE
□	8600 LB	130.2 (FWD)	4080 FT
○	9500 LB	137.1 (MID)	4250 FT
△	9500 LB	134.0 (FWD)	3750 FT
○	8300 LB	136.2	4840 FT
△	7000 LB	130.0 (FWD)	5400 FT
○	8600 LB	130.2 (FWD)	12,760 FT
◇	8400 LB	144.0 (AFT)	4780 FT
▽	9500 LB	143.9 (AFT)	5700 FT

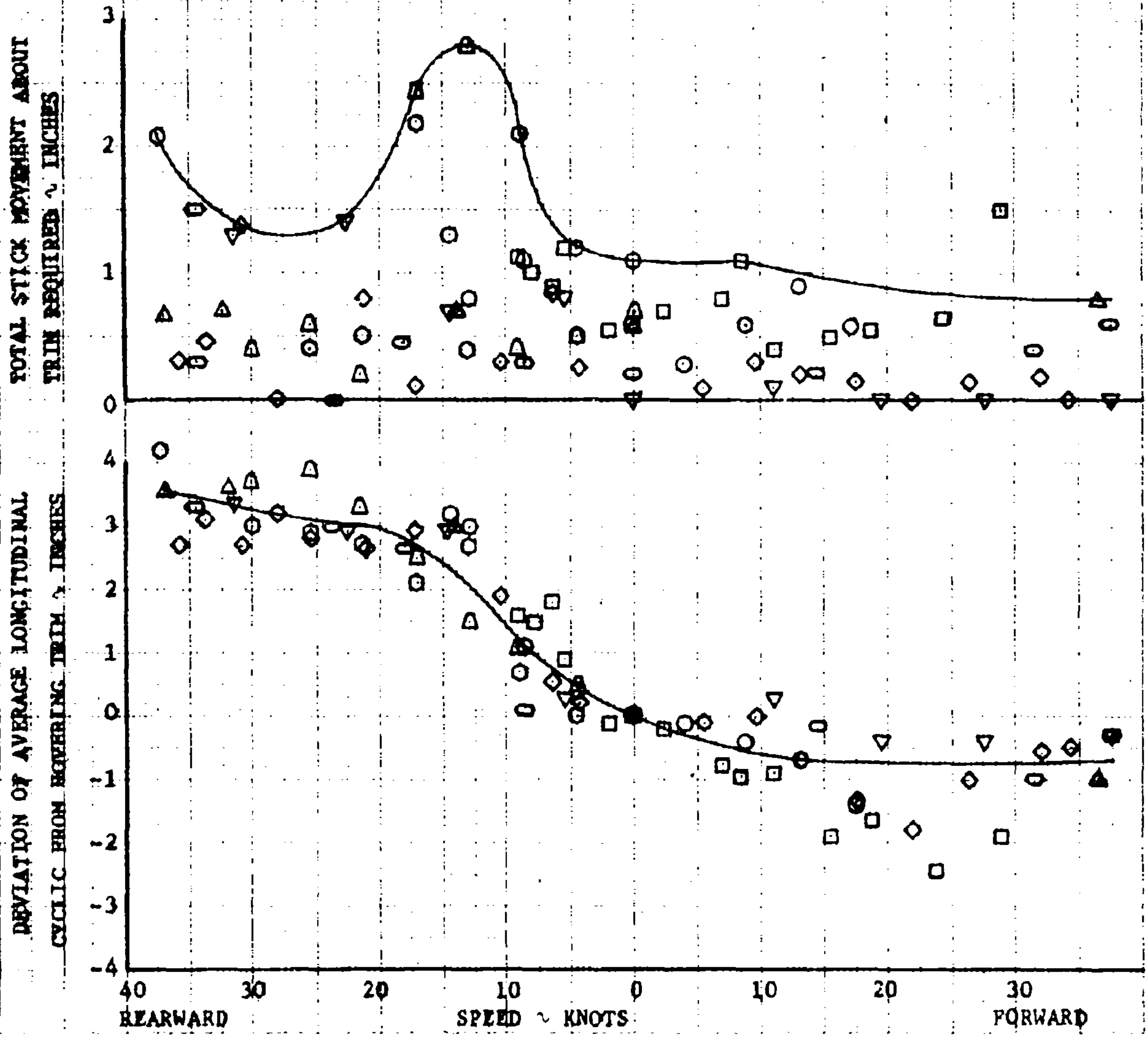


FIGURE 87

SIDEWARD FLIGHT

YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM | SKID HEIGHT = 10 to 20 FT
DENSITY ALTITUDE = 4080 FT | WIND < 3 KNOTS, CORRECTED TO ZERO
LONGITUDINAL CG = 130.2 (FWD) | GROSS WEIGHT = 8600 LB
LATERAL CG = 0.0 (MID)

NOTES:

1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

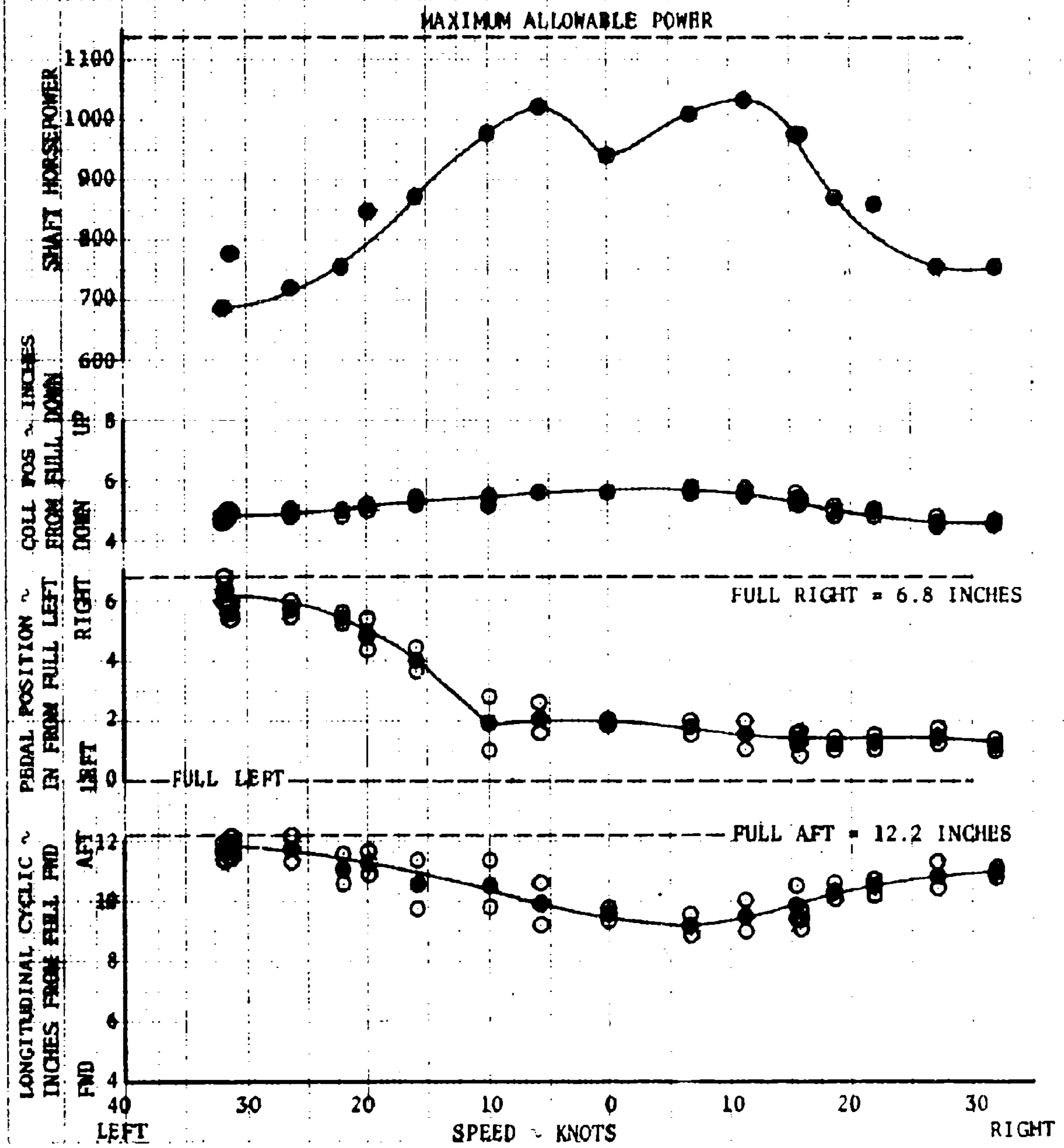


FIGURE [illegible]
REARWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 4080 FT
LONGITUDINAL CG = 130.2 (FWD)
LATERAL CG = 0.0 (MID)

SKID HEIGHT = 10 to 20 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 8600 LB

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

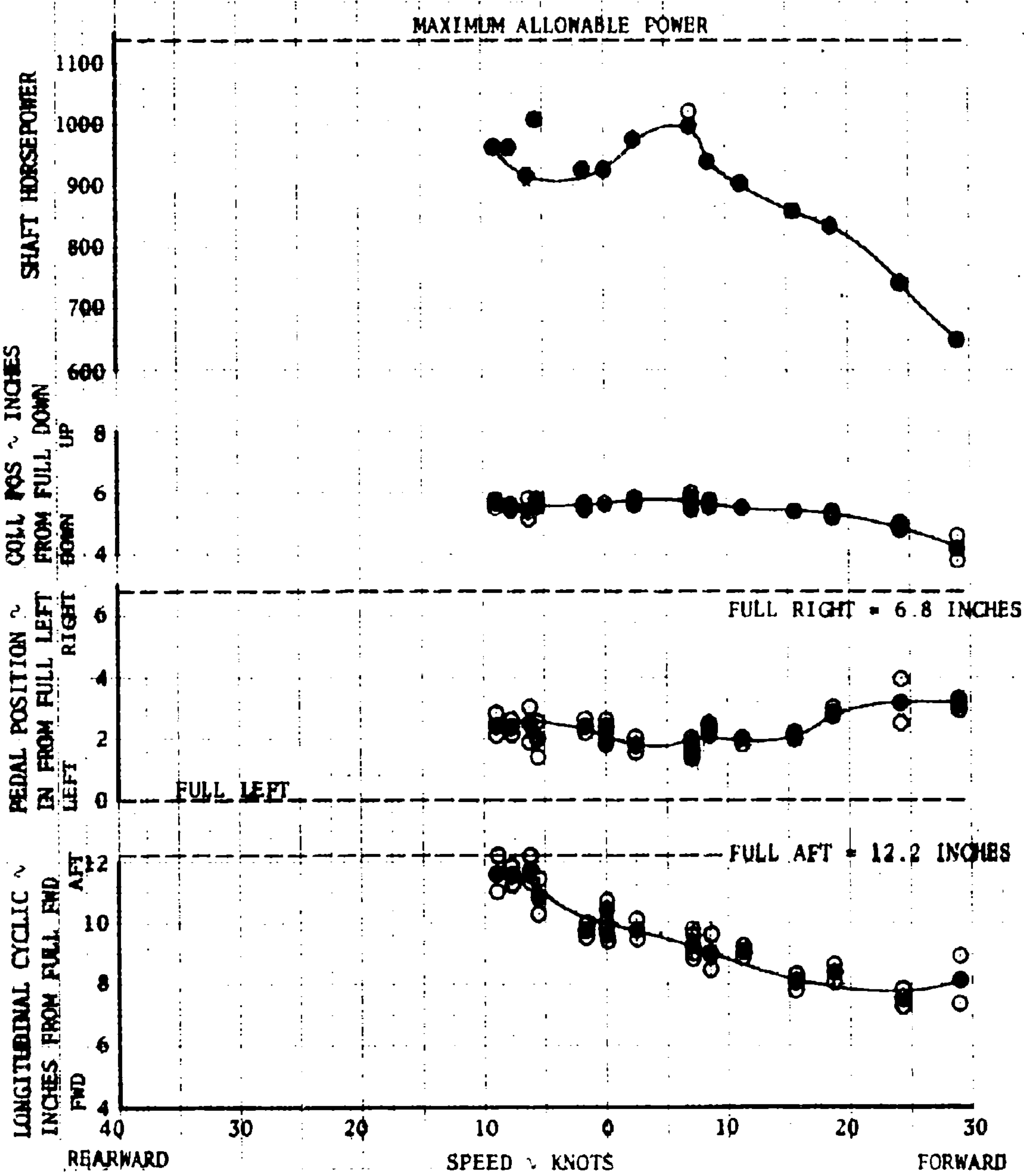


FIGURE 89
SIDEWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 4250 FT
LONGITUDINAL CG = 137.1 (MID)
LATERAL CG = +2.5 (RIGHT)

SKID HEIGHT = 10 to 20 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 9500 LB

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

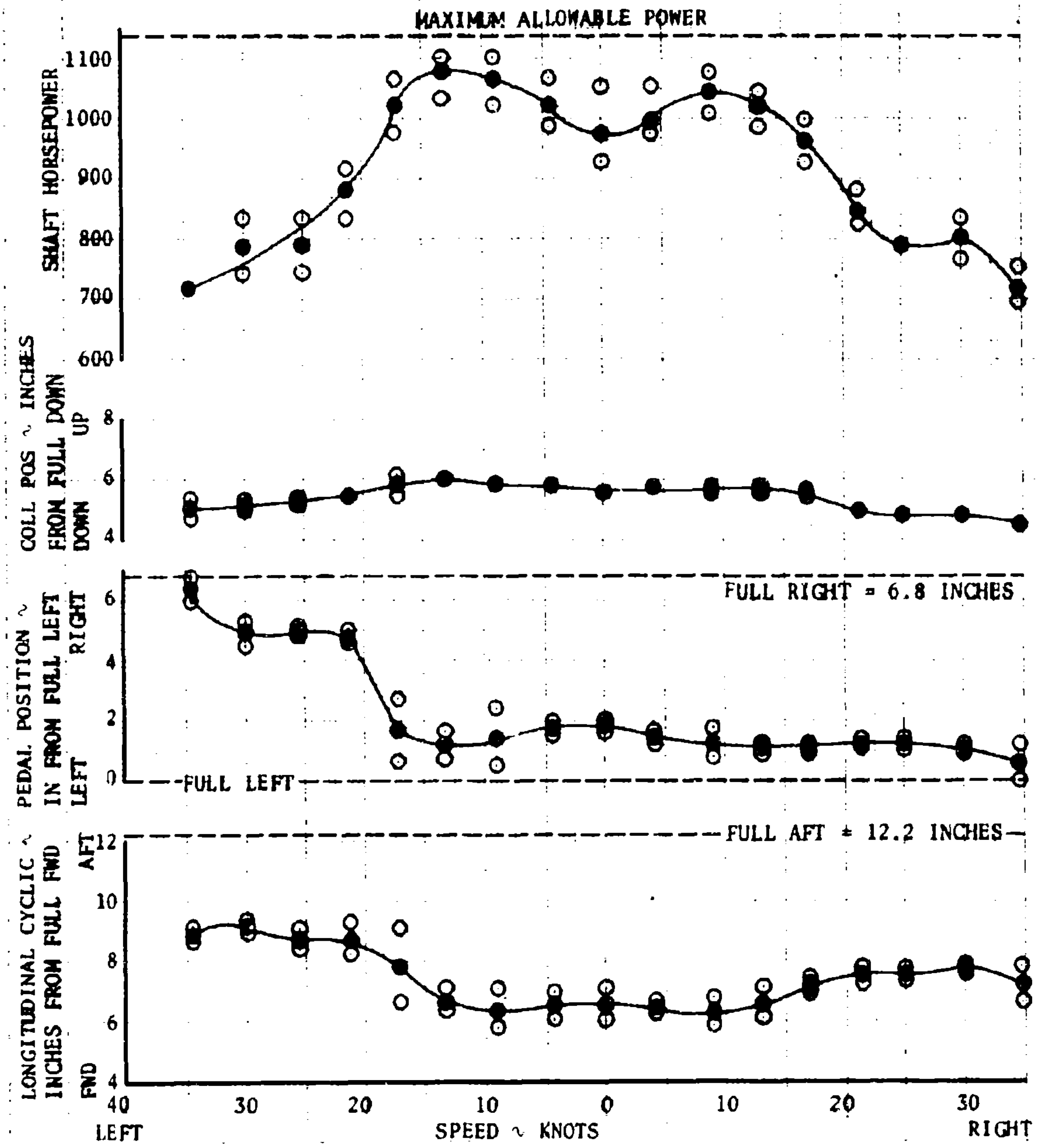


FIGURE 90
REARWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 4250 FT
LONGITUDINAL CG = 137.1 (MID)
LATERAL CG = +2.5 (RIGHT)
SKID HEIGHT = 10 to 20 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 9500 LB

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

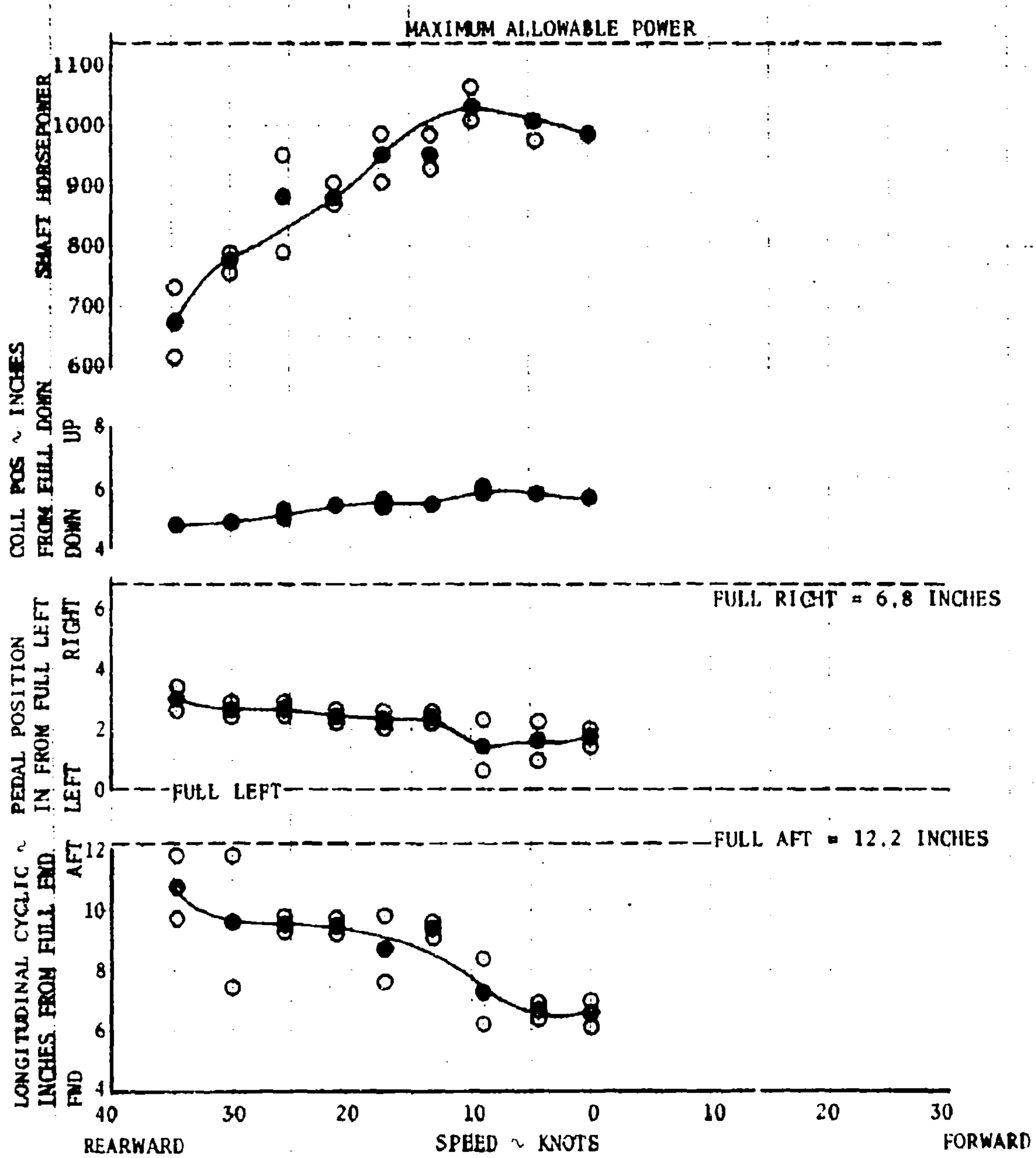


FIGURE 91
SIDEWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 3750 FT
LONGITUDINAL CG = 134.0 (FWD)
LATERAL CG = -2.0 (LEFT)
SKID HEIGHT = 10 to 20 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 9500 LB

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

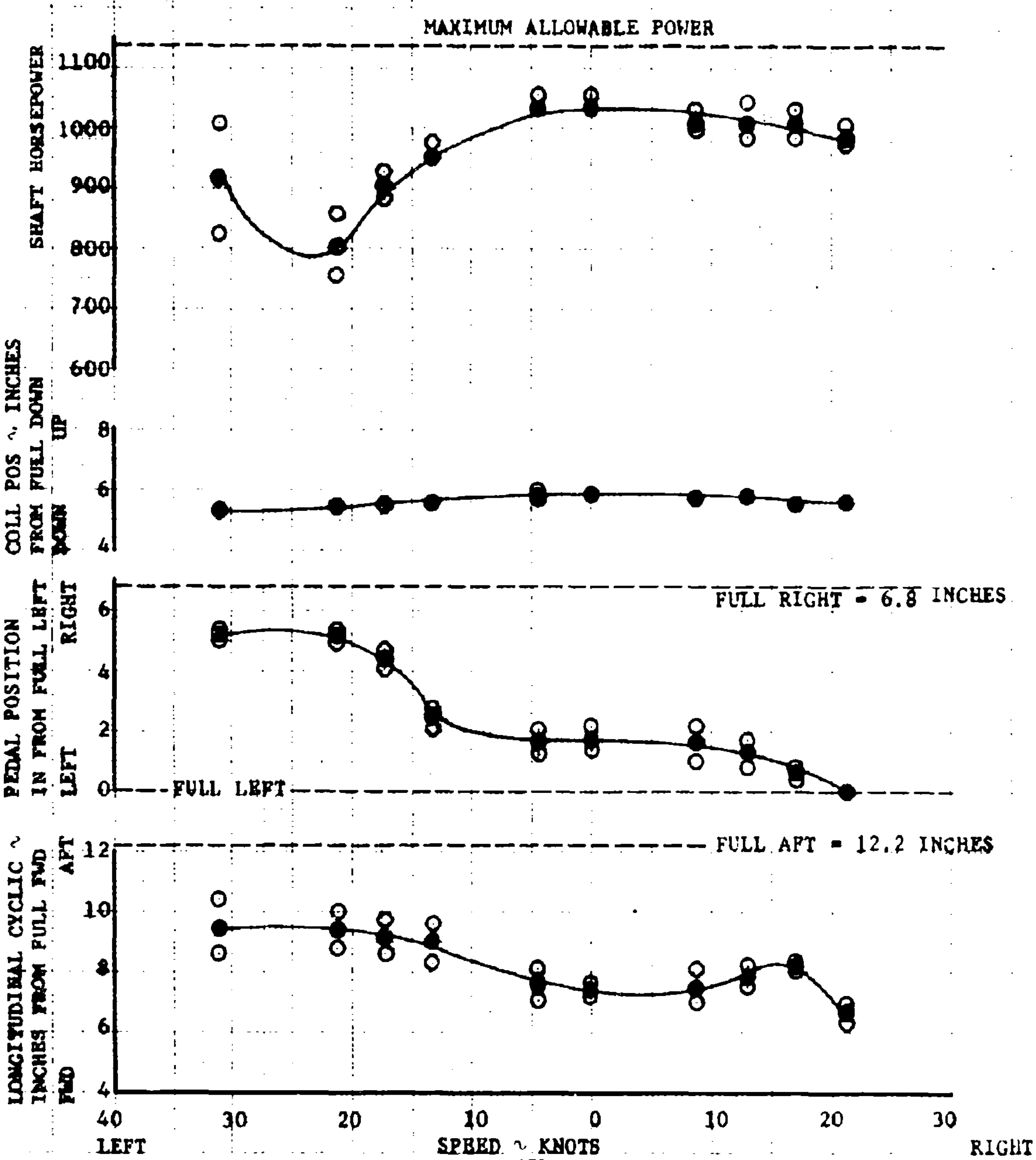


FIGURE 92
REARWARD FLIGHT
YUH-1H S/N 60-6028

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 3750 FT
LONGITUDINAL CG = 134.0 (FWD)
LATERAL CG = -2.0 (LEFT)
SKID HEIGHT = 10 to 20 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 9500 LB

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

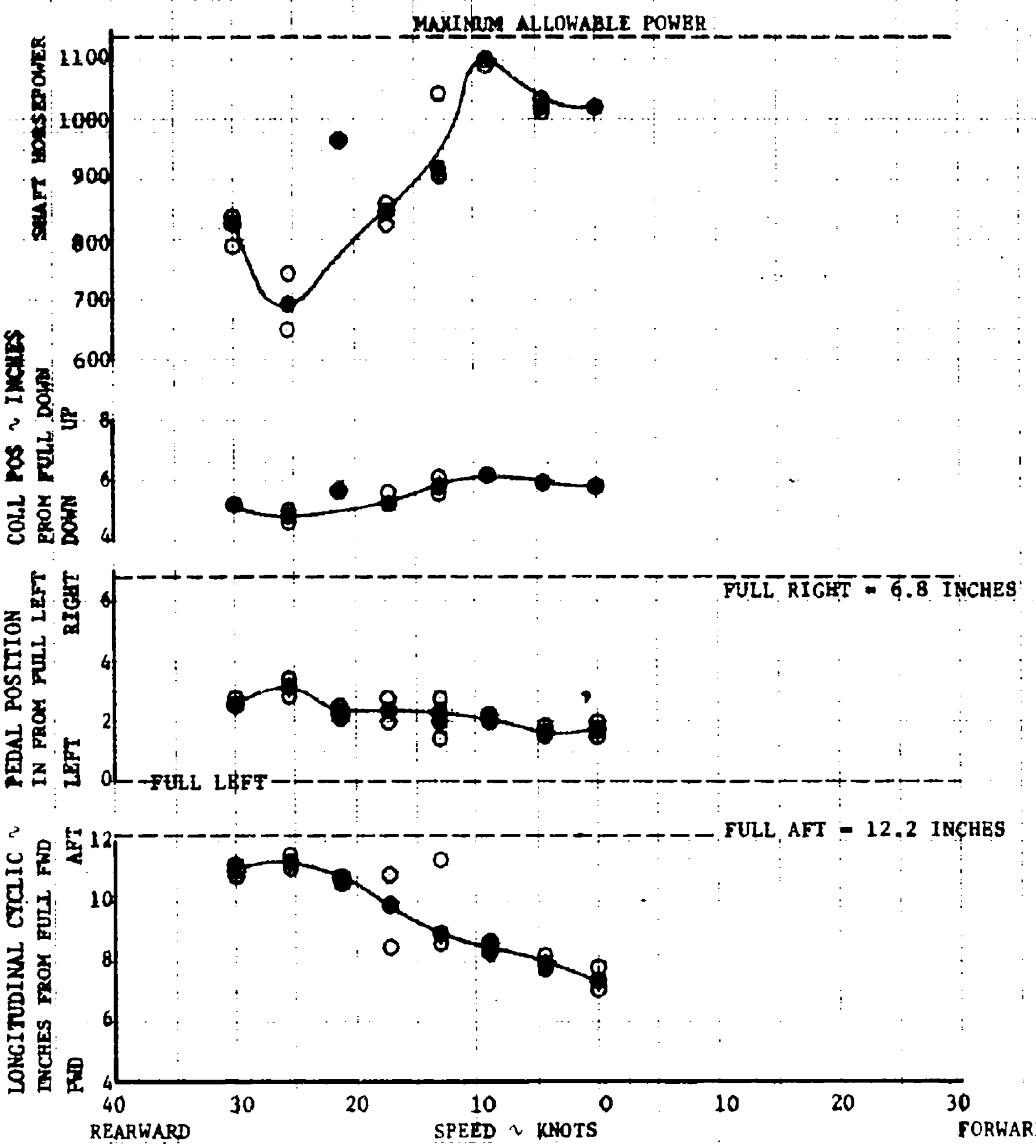


FIGURE 93
SIDEWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM SKID HEIGHT = 10 to 20 FT
DENSITY ALTITUDE = 4840 FT WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 8300 LB
LONG CG = □ 130.0, ▽ 133.1, ○ 136.2 LAT CG = 0.0 (MID)

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

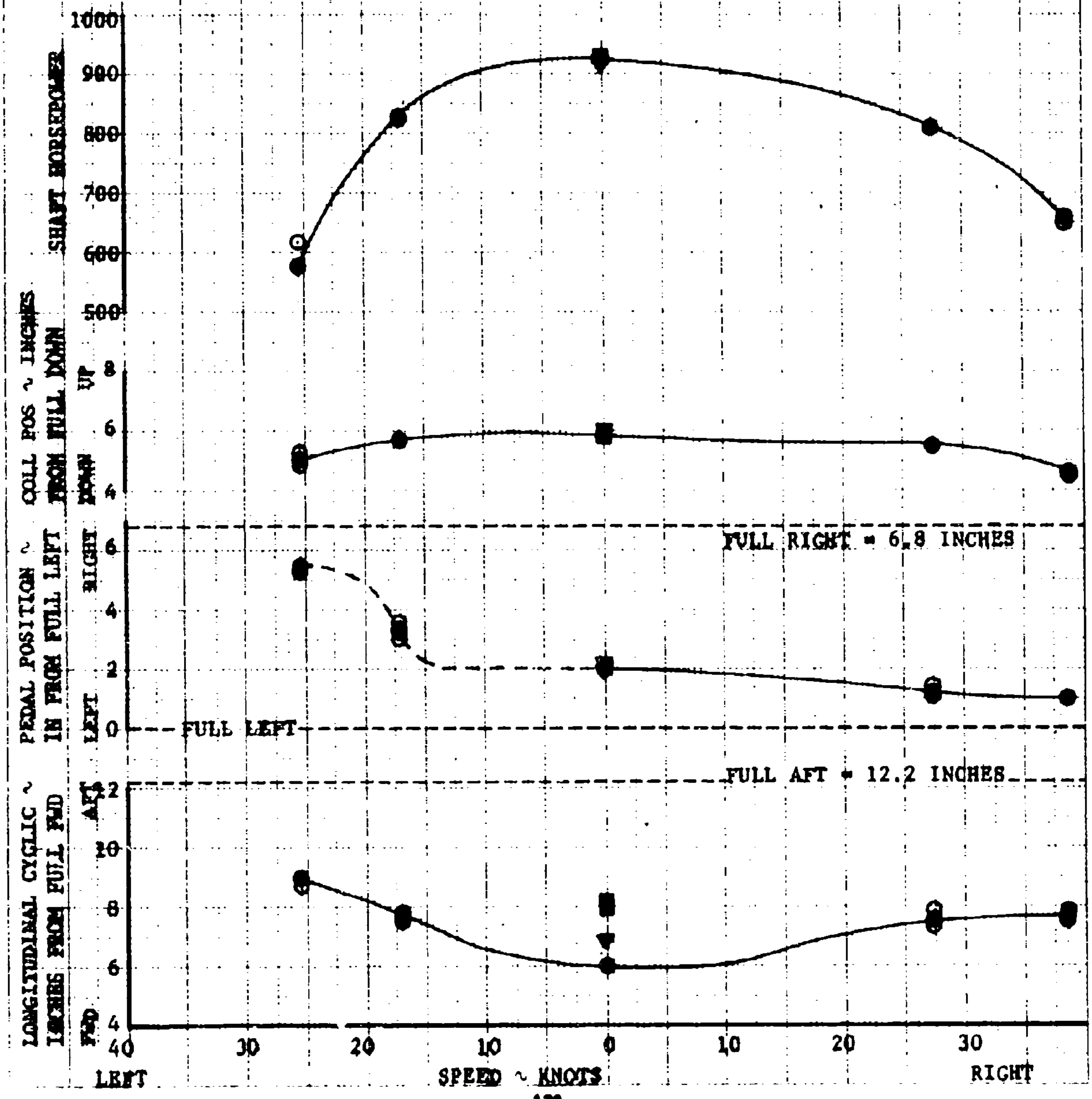


FIGURE 9A
REARWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM SKID HEIGHT = 10 to 20 FT
DENSITY ALTITUDE = 4840 FT WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 8300 LB
LONG CG = ◻ 130.0, ▽ 133.1, ○ 136.2 LAT CG = 0.0 (MID)

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

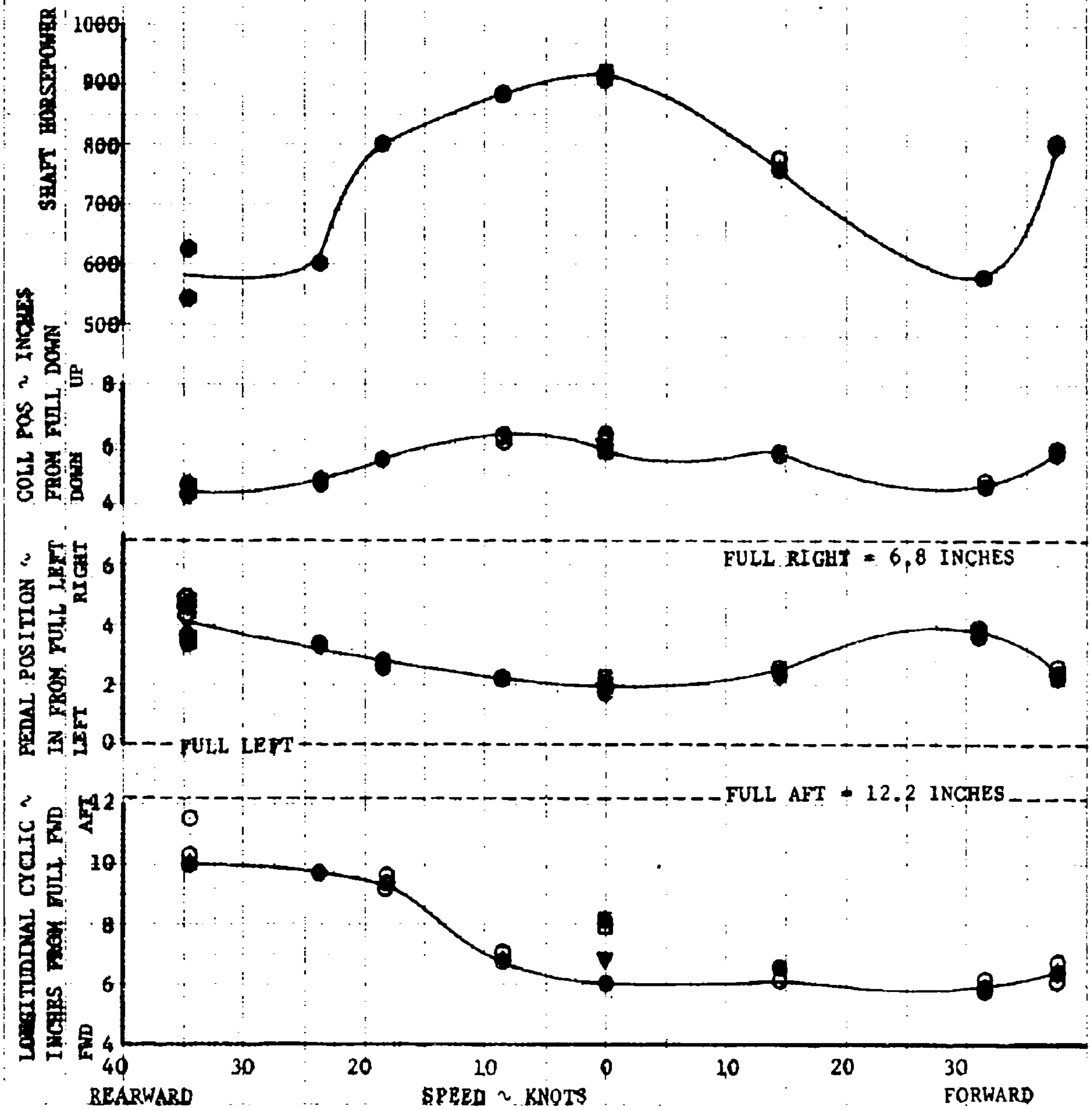


FIGURE 95
SIDEWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM SKID HEIGHT = 10 to 20 FT
DENSITY ALTITUDE = 5400 FT WIND < 3 KNOTS, CORRECTED TO ZERO
LONGITUDINAL CG = 130.0 (FWD) LATERAL CG = 0.0 (MID)
GROSS WEIGHT = ○ 7000, ▽ 7600, □ 8600

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

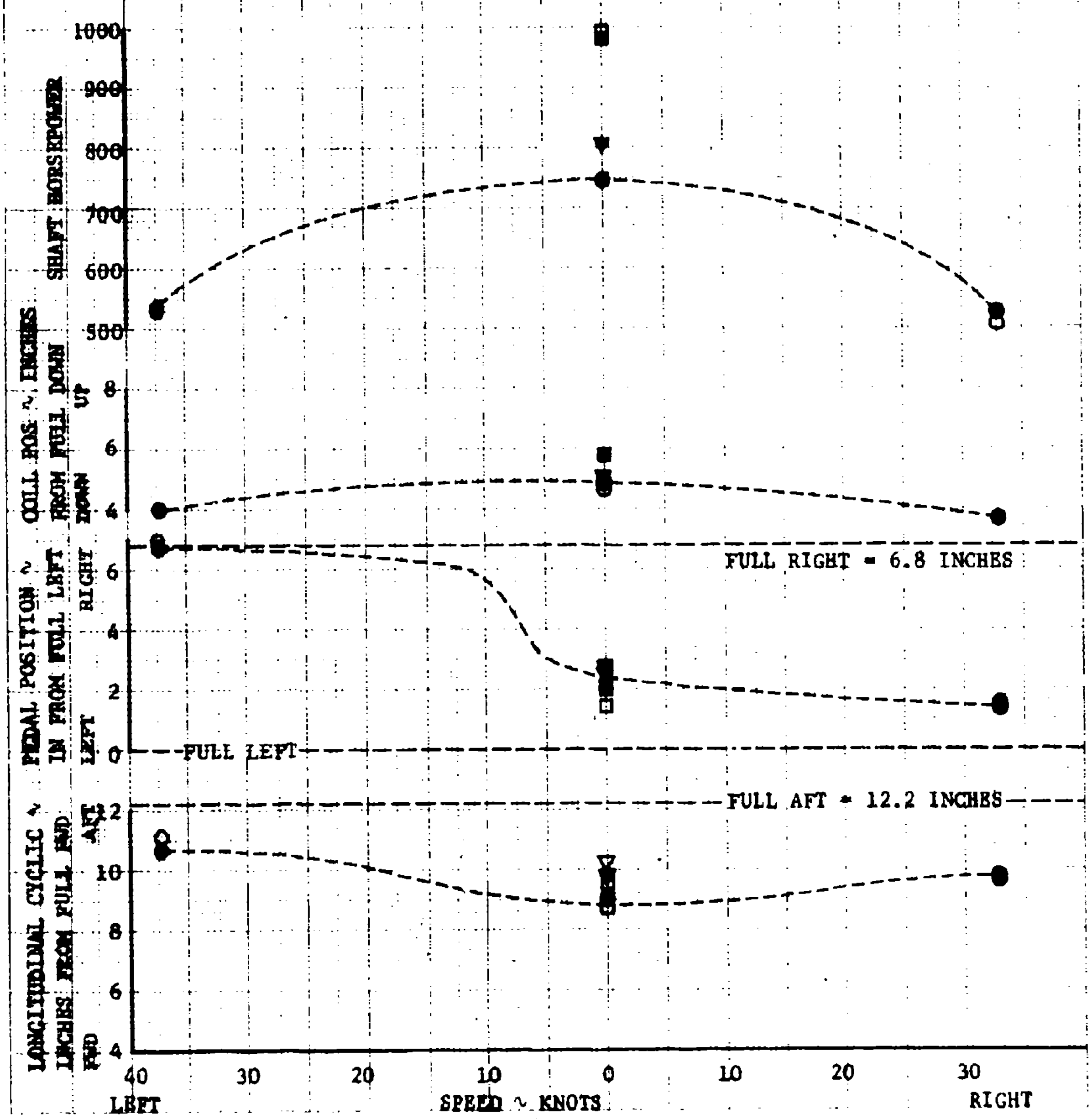


FIGURE 96
REARWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM SKID HEIGHT = 10 to 20 FT
DENSITY ALTITUDE = 5400 FT WIND < 3 KNOTS, CORRECTED TO ZERO
LONGITUDINAL CG = 130.0 (FWD) LATERAL CG = 0.0 (MID)
GROSS WEIGHT = ○ 7000, ▽ 7600, □ 8600

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

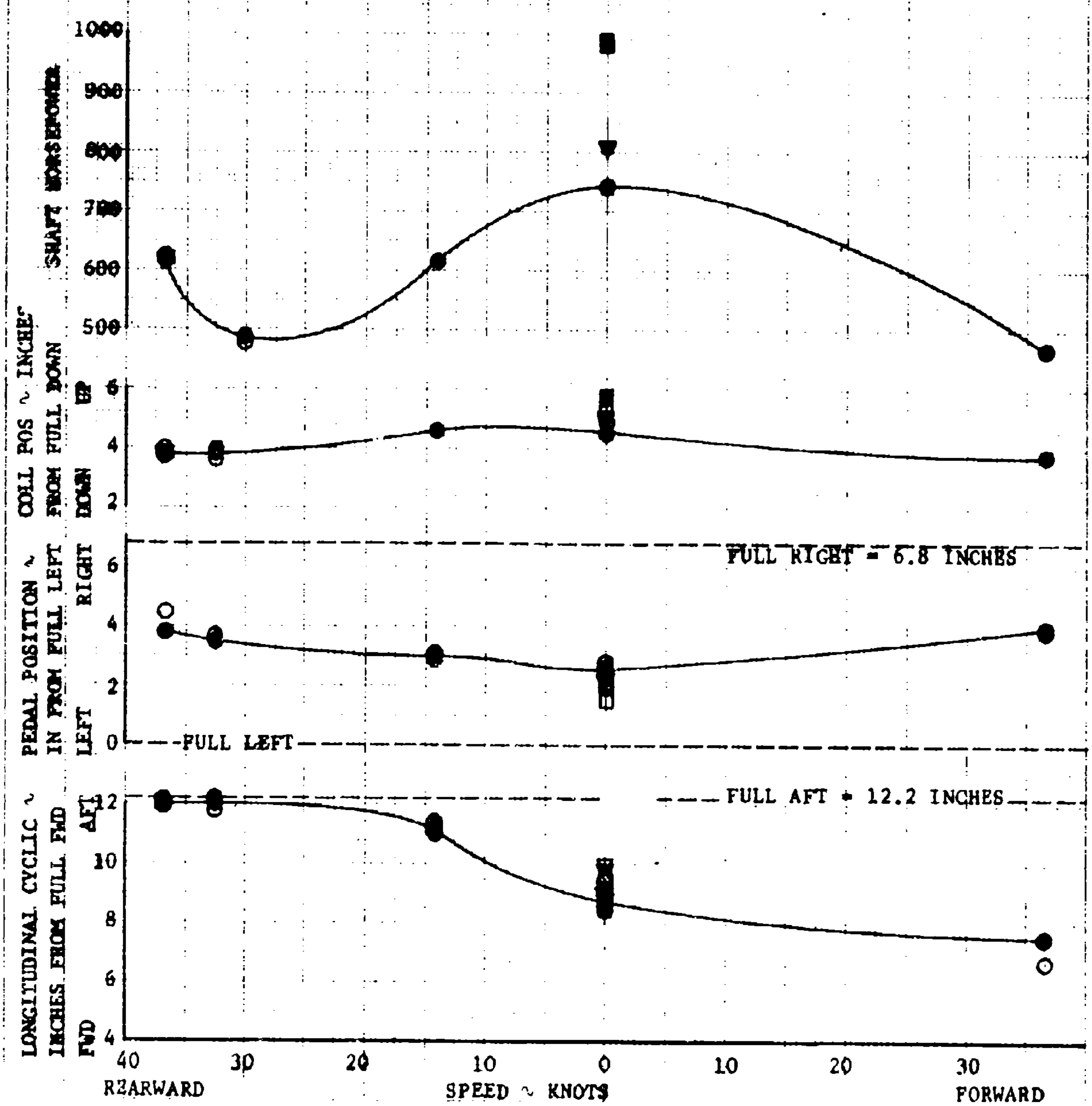


FIGURE 97
SIDEWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 12,760 FT
LONGITUDINAL CG = 130.2 (FWD)
LATERAL CG = 0.0 (MID)

SKID HEIGHT = 10 to 20 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 8600 LB

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

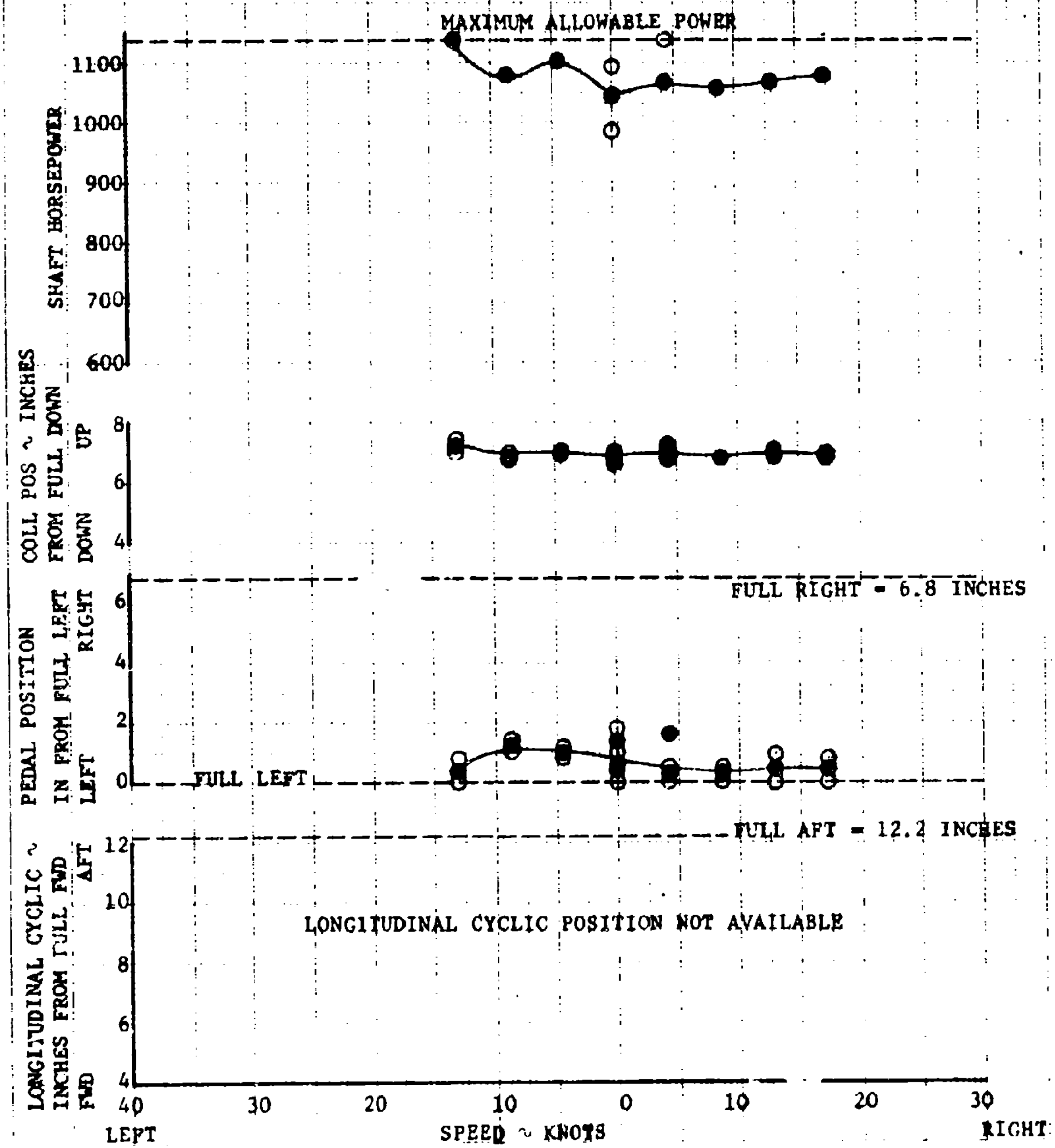


FIGURE 98
REARWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM
SKID HEIGHT = 10 to 20 FT
DENSITY ALTITUDE = 12,760 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
LONGITUDINAL CG = 130.2 (FWD)
GROSS WEIGHT = 8600 LB
LATERAL CG = 0.0 (MID)

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

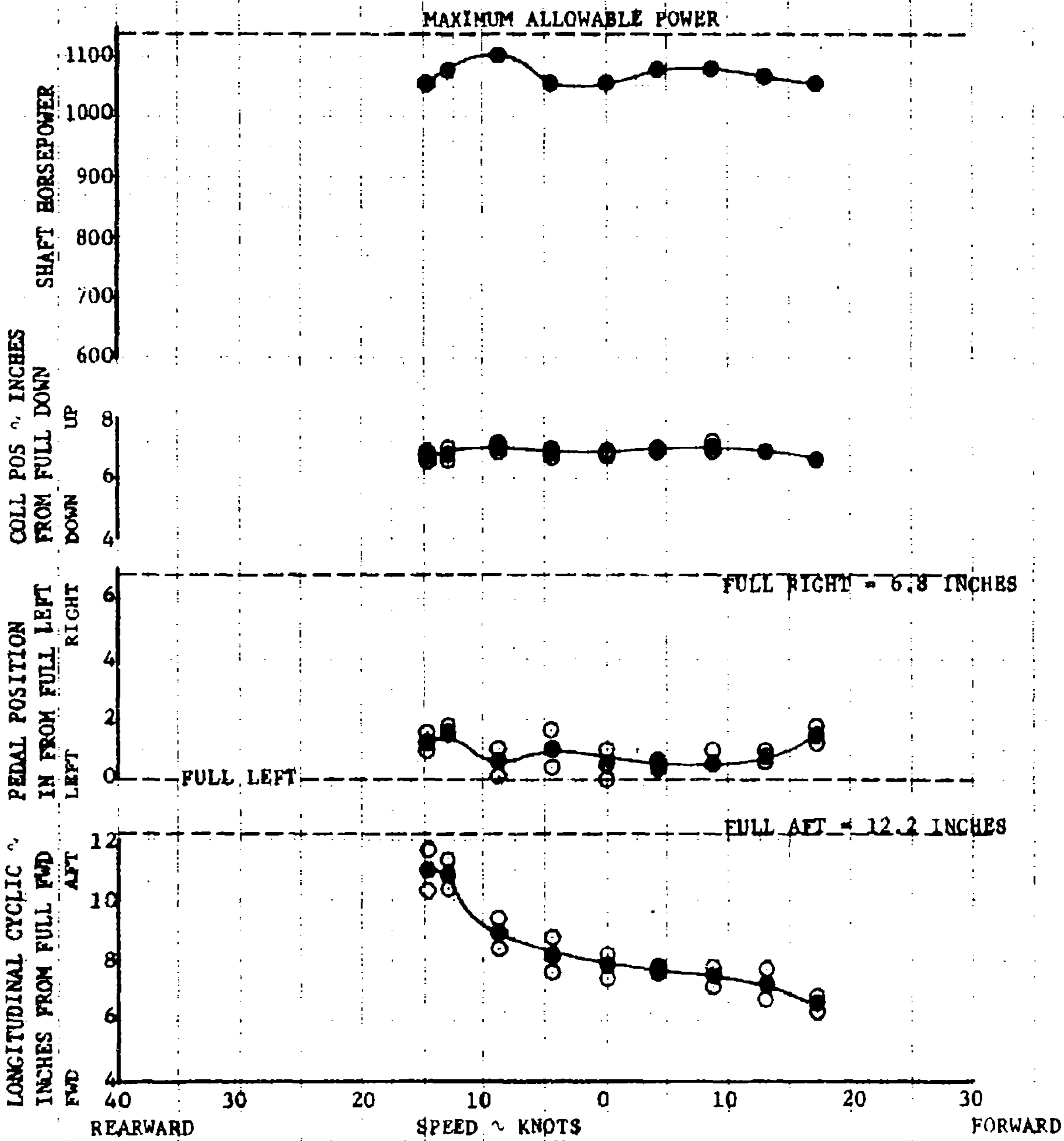


FIGURE 99
SIDEWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE =4780 FT
LONGITUDINAL CG = 144.0 (AFT)
LATERAL CG = 0.0 (MID)

SKID HEIGHT = 10 to 20 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 8600 LB

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

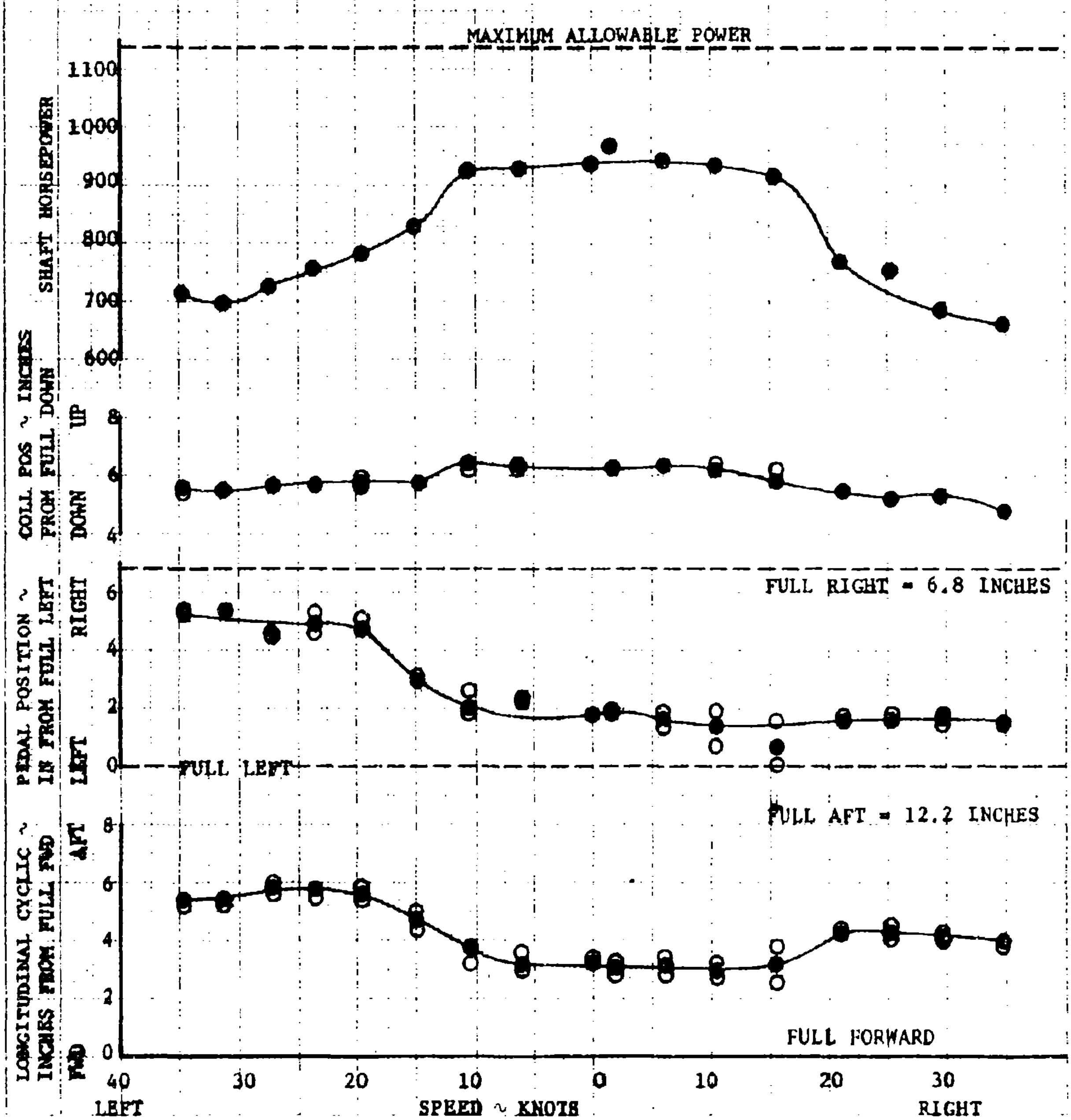


FIGURE 100
REARWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 4780 FT
LONGITUDINAL CG = 144.0 (AFT)
LATERAL CG = 0.0 (MID)

SKID HEIGHT = 10 to 20 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 8600 LB

NOTES:

1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

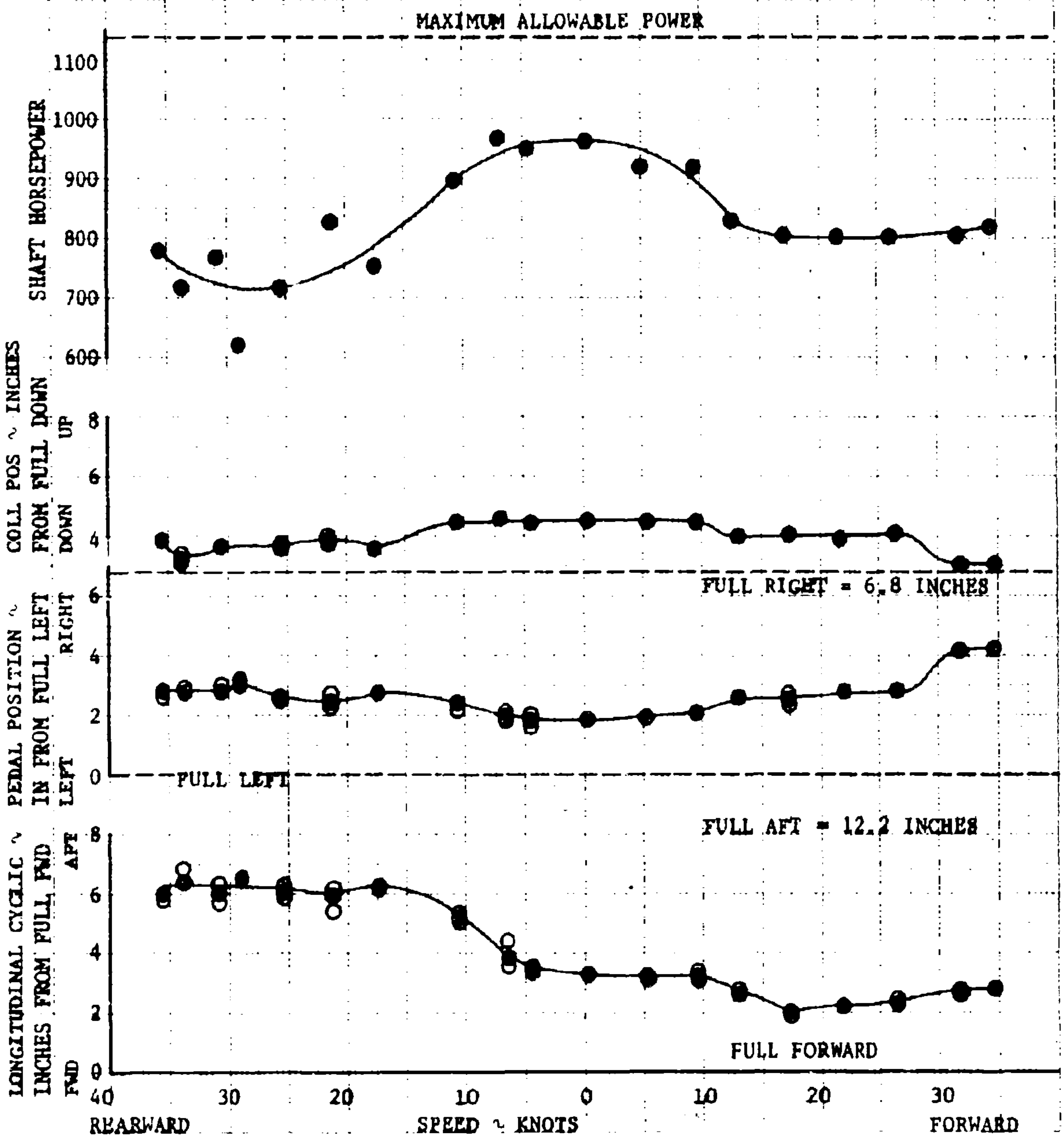


FIGURE 181
SIDEWARD FLIGHT
YUH-1H S/N 60-6029

ROTOR SPEED = 324 RPM | SKID HEIGHT = 10 to 20 FT
DENSITY ALTITUDE = 5700 FT | WIND < 3 KNOTS, CORRECTED TO ZERO
LONGITUDINAL CG = 143.9 (AFT) | GROSS WEIGHT = 9500 LB
LATERAL CG = 0.0 (MID)

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight

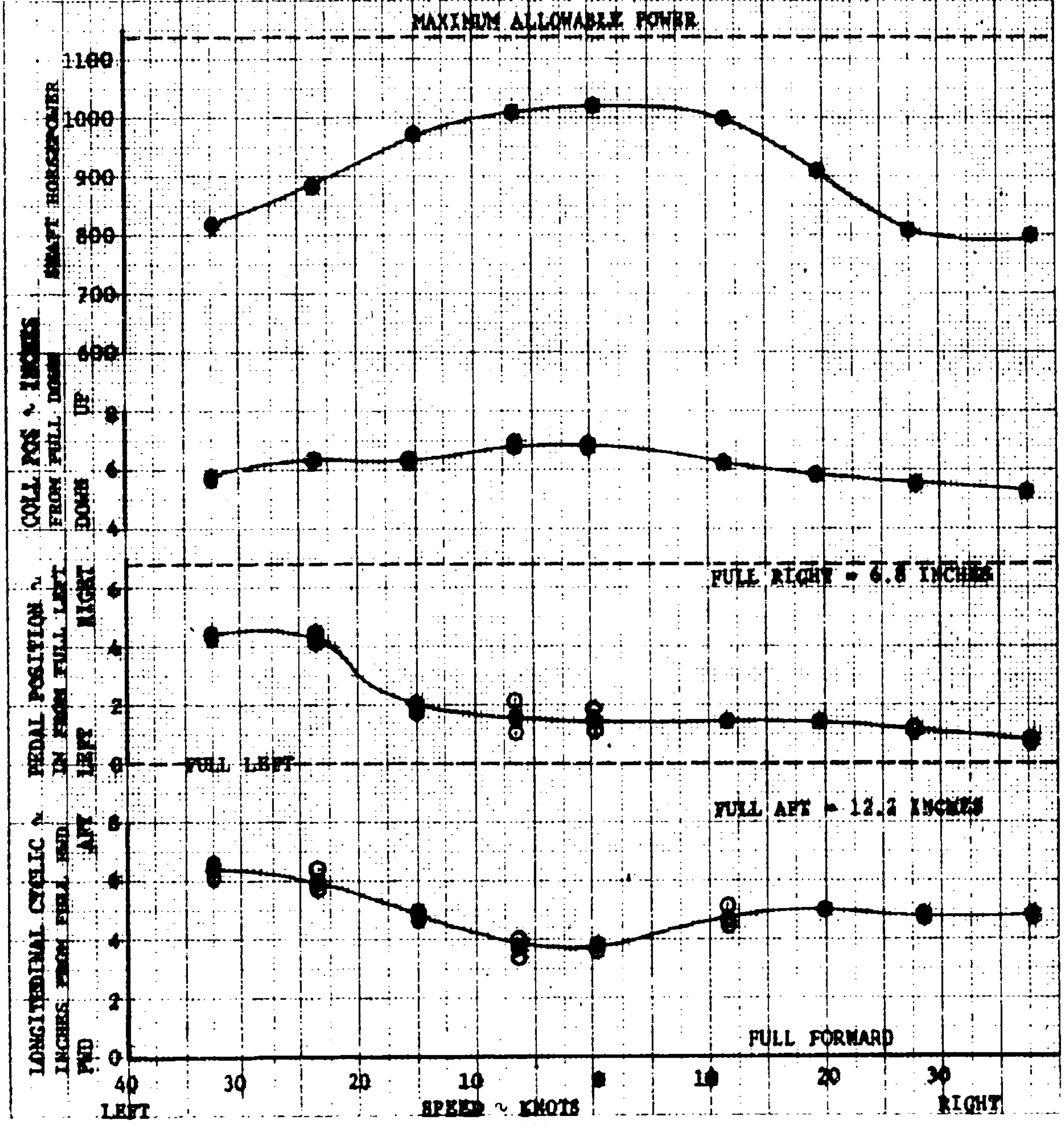
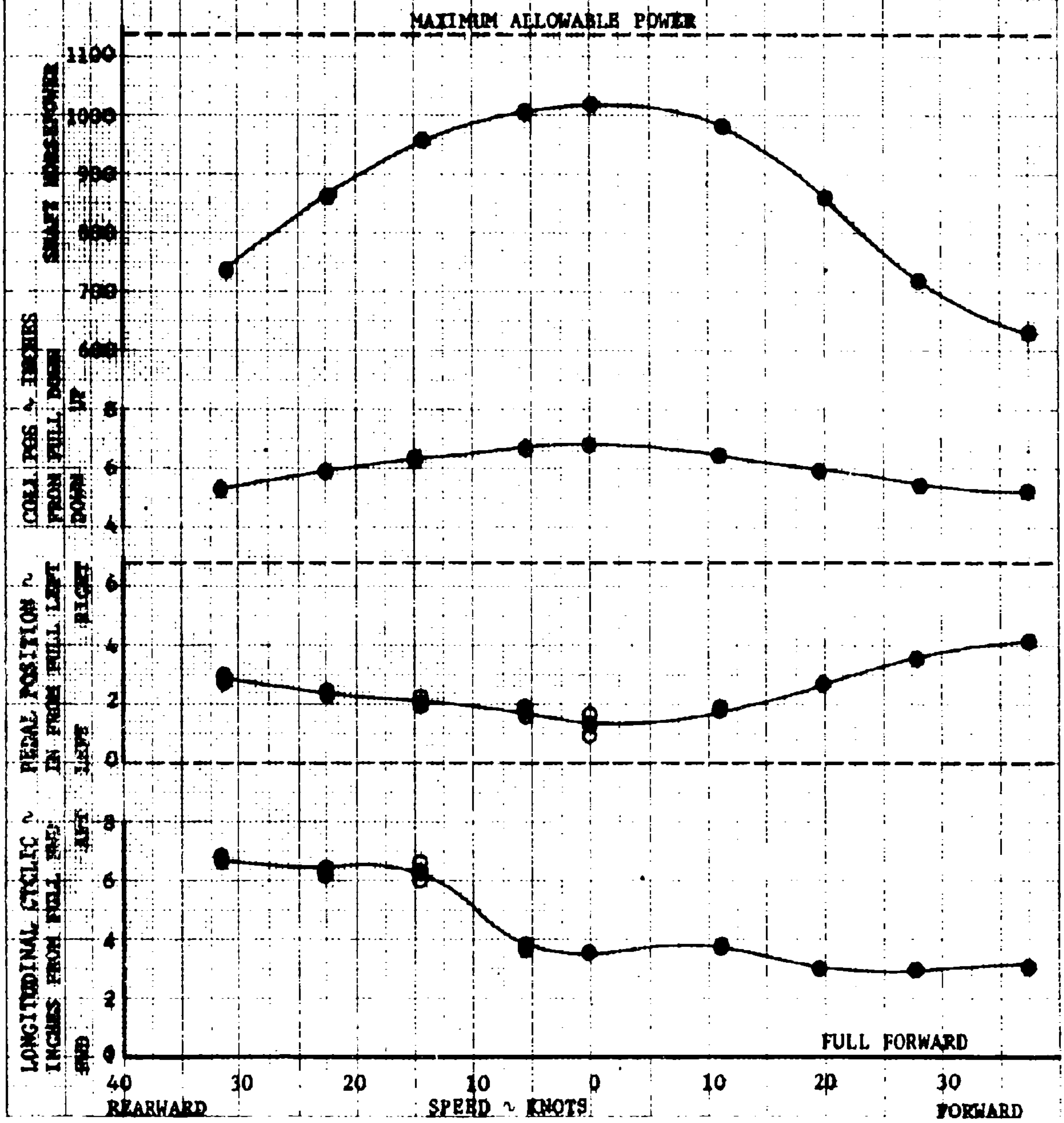


FIGURE 102
REARWARD FLIGHT
YUH-1M S/N 60-6029

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 5700 FT
LONGITUDINAL CG = 143.9 (AFT)
LATERAL CG = 0.0 (MID)
SKID HEIGHT = 10 to 20 FT
WIND < 3 KNOTS, CORRECTED TO ZERO
GROSS WEIGHT = 9500 LB

NOTES:
1. Solid Symbols Denote Average Conditions During 5-10 Seconds of Stable Flight
2. Open Symbols Denote Extreme Conditions During 5-10 Seconds of Stable Flight



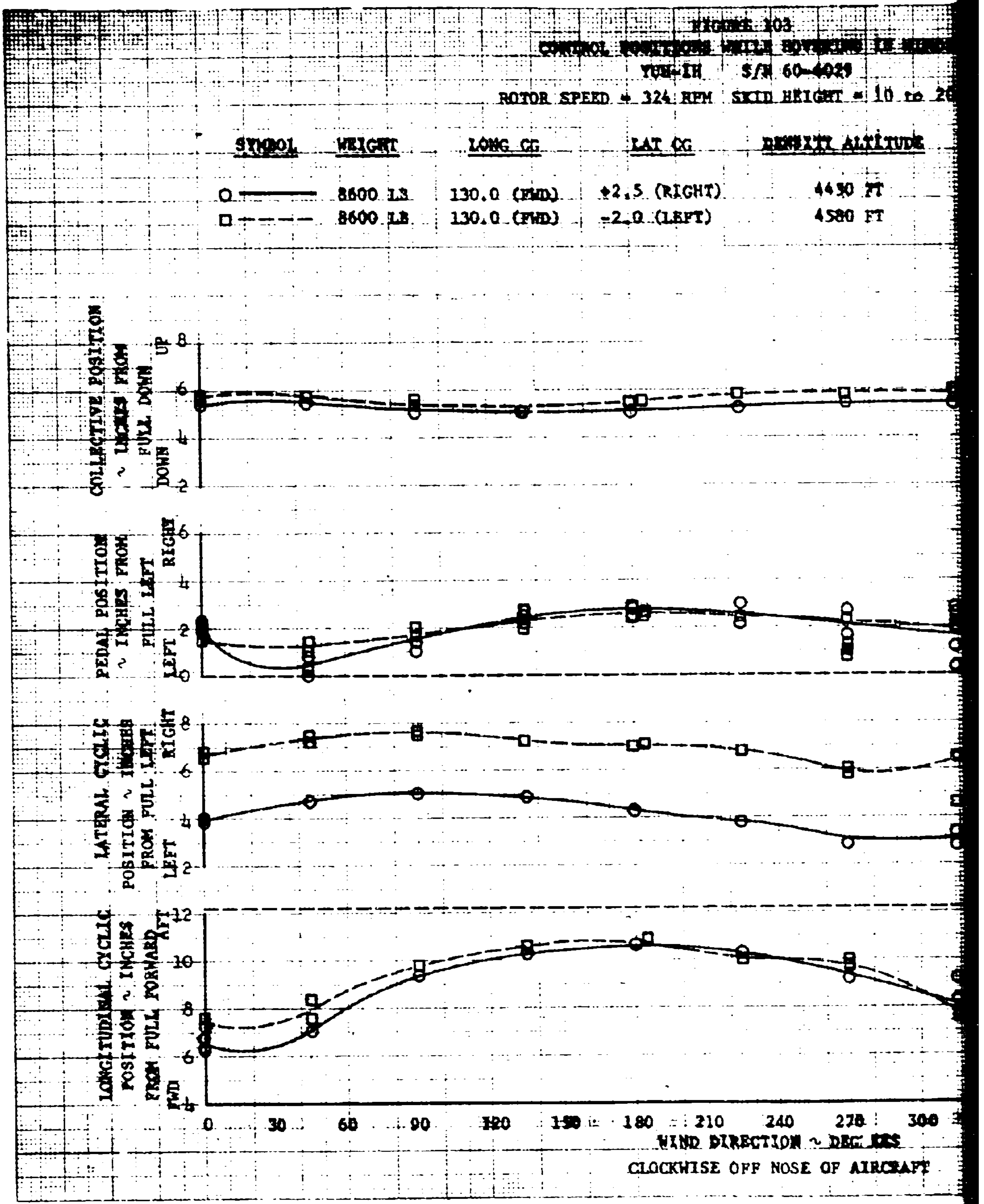
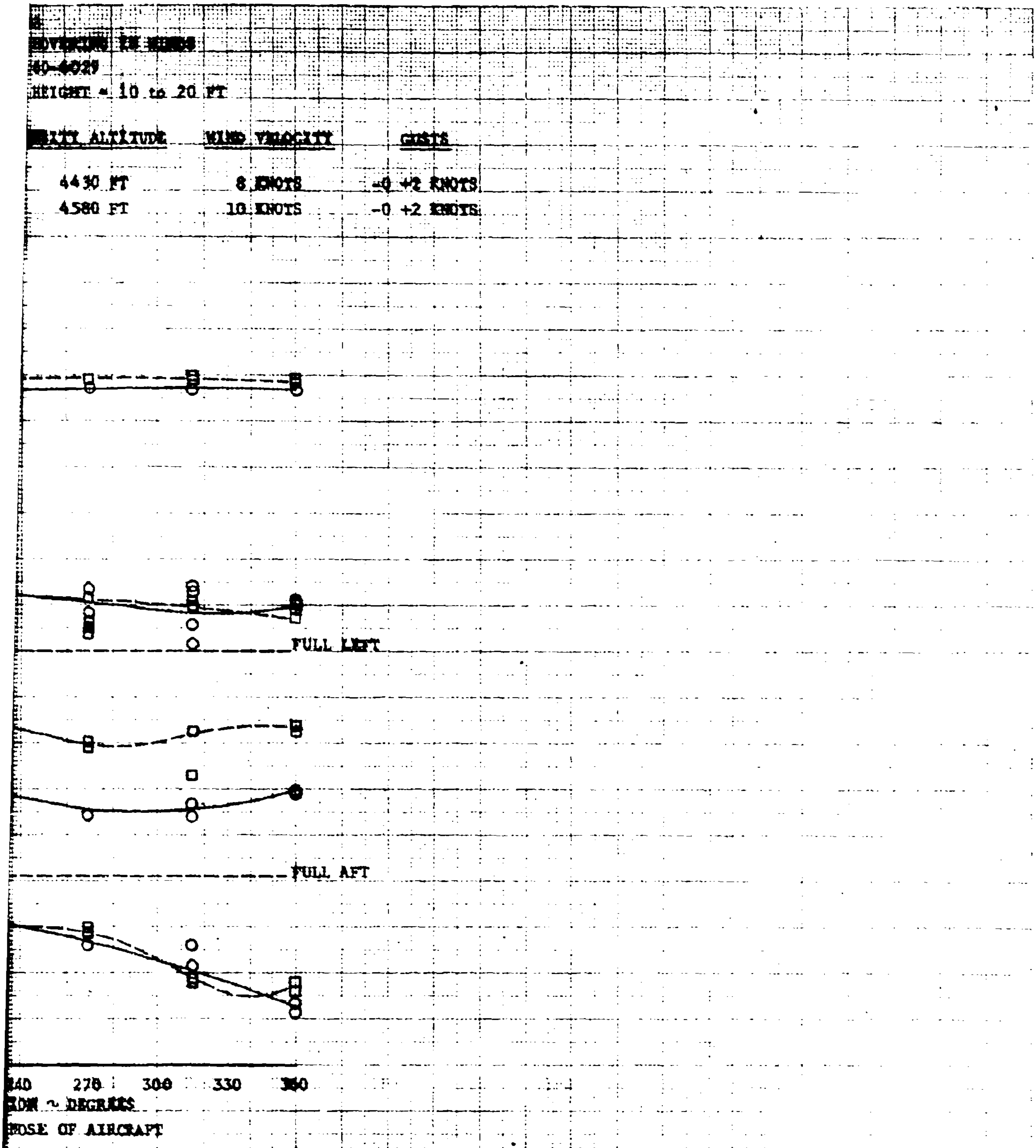


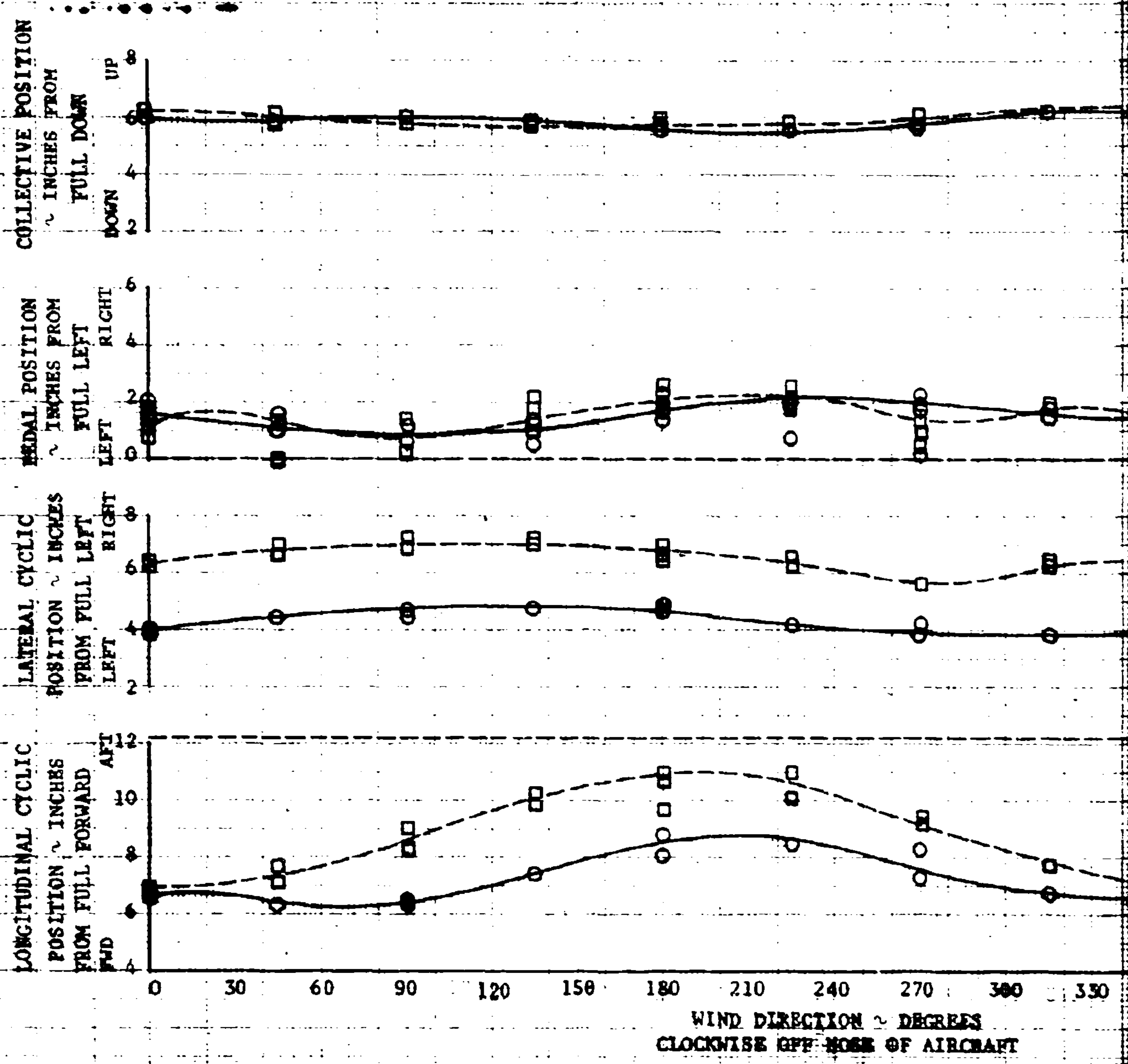
FIGURE 103
YUH-1H S/N 60-4029
ROTOR SPEED = 324 RPM SKID HEIGHT = 10 to 2
SYMBOL WEIGHT LONG CG LAT CG DENSITY ALTITUDE
8600 LB 130.0 (FWD) +2.5 (RIGHT) 4430 FT
8600 LB 130.0 (FWD) -2.0 (LEFT) 4580 FT
COLLECTIVE POSITION ~ INCHES FROM FULL DOWN
UP
DOWN
PEDAL POSITION ~ INCHES FROM FULL LEFT
RIGHT
LEFT
LATERAL CYCLIC POSITION ~ INCHES FROM FULL LEFT
RIGHT
LEFT
LONGITUDINAL CYCLIC POSITION ~ INCHES FROM FULL FORWARD
AFT
FWD
0 30 60 90 120 150 180 210 240 270 300
WIND DIRECTION ~ DEGREES
CLOCKWISE OFF NOSE OF AIRCRAFT


HEIGHT ≈ 10 to 20 FT
ALTITUDE
WIND VELOCITY
GUSTS
4430 FT
8 KNOTS
-0 +2 KNOTS
4580 FT
10 KNOTS
-0 +2 KNOTS
FULL LEFT
FULL AFT
270
300
330
360
DEGREES
NOSE OF AIRCRAFT

2

FIGURE 104
CONTROL POSITIONS WHILE HOVERING IN WINDS
YUH-1H S/N 60-6029
ROTOR SPEED = 324 RPM SKID HEIGHT = 10 to 20 FT

SYMBOL	WEIGHT	LONG CG	LAT CG	DENSITY ALTITUDE	WIND
○ ———	9450 LB	134.0 (FWD)	+2.5 (RIGHT)	4570 FT	9
□ – – –	9400 LB	134.0 (FWD)	-2.0 (LEFT)	4950 FT	10



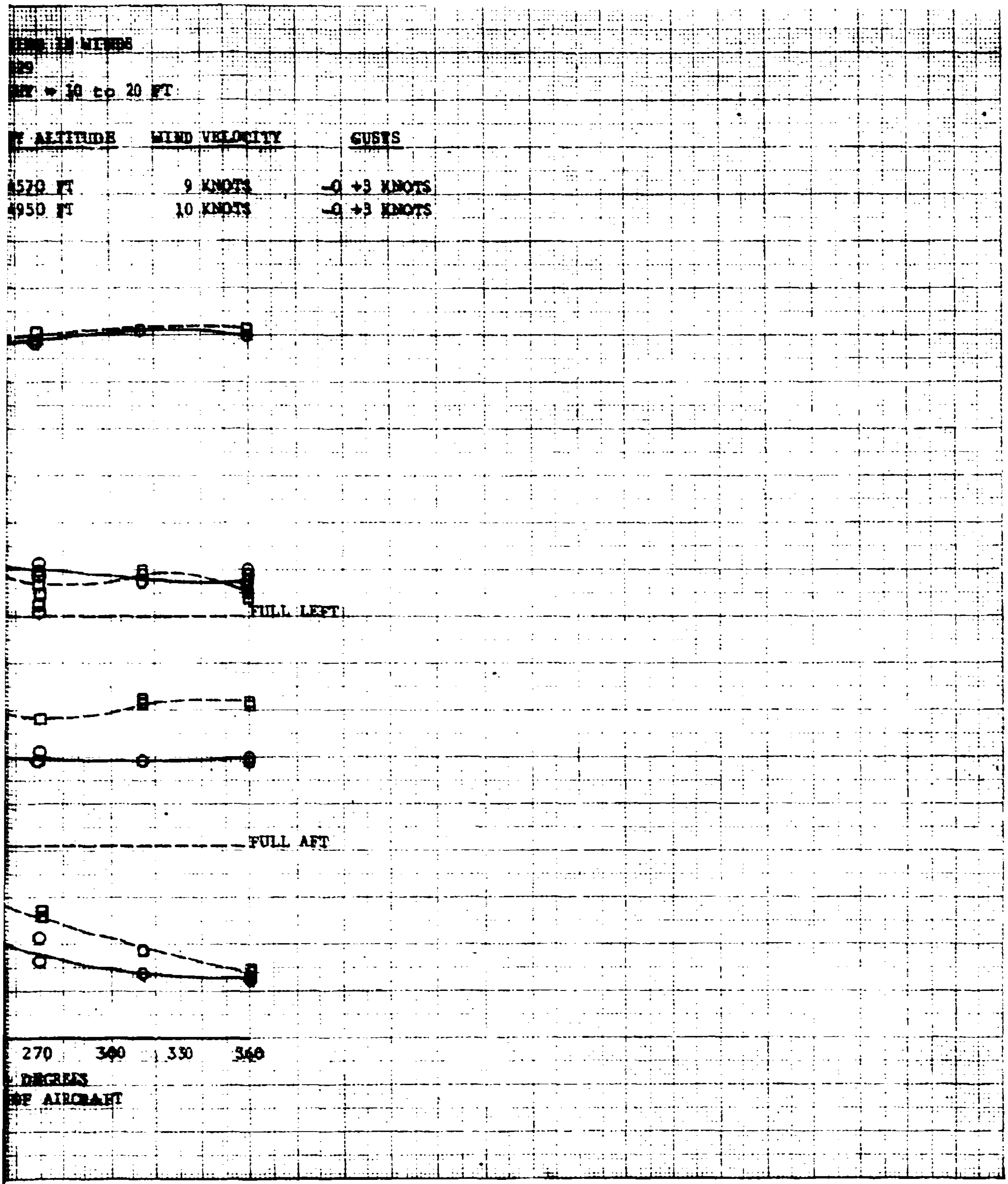

ALTITUDE
WIND VELOCITY
GUSTS
4570 FT
9 KNOTS
-0 +3 KNOTS
4950 FT
10 KNOTS
-0 +3 KNOTS
FULL LEFT
FULL AFT
270
300
330
360
DEGREES
AIRCRAFT

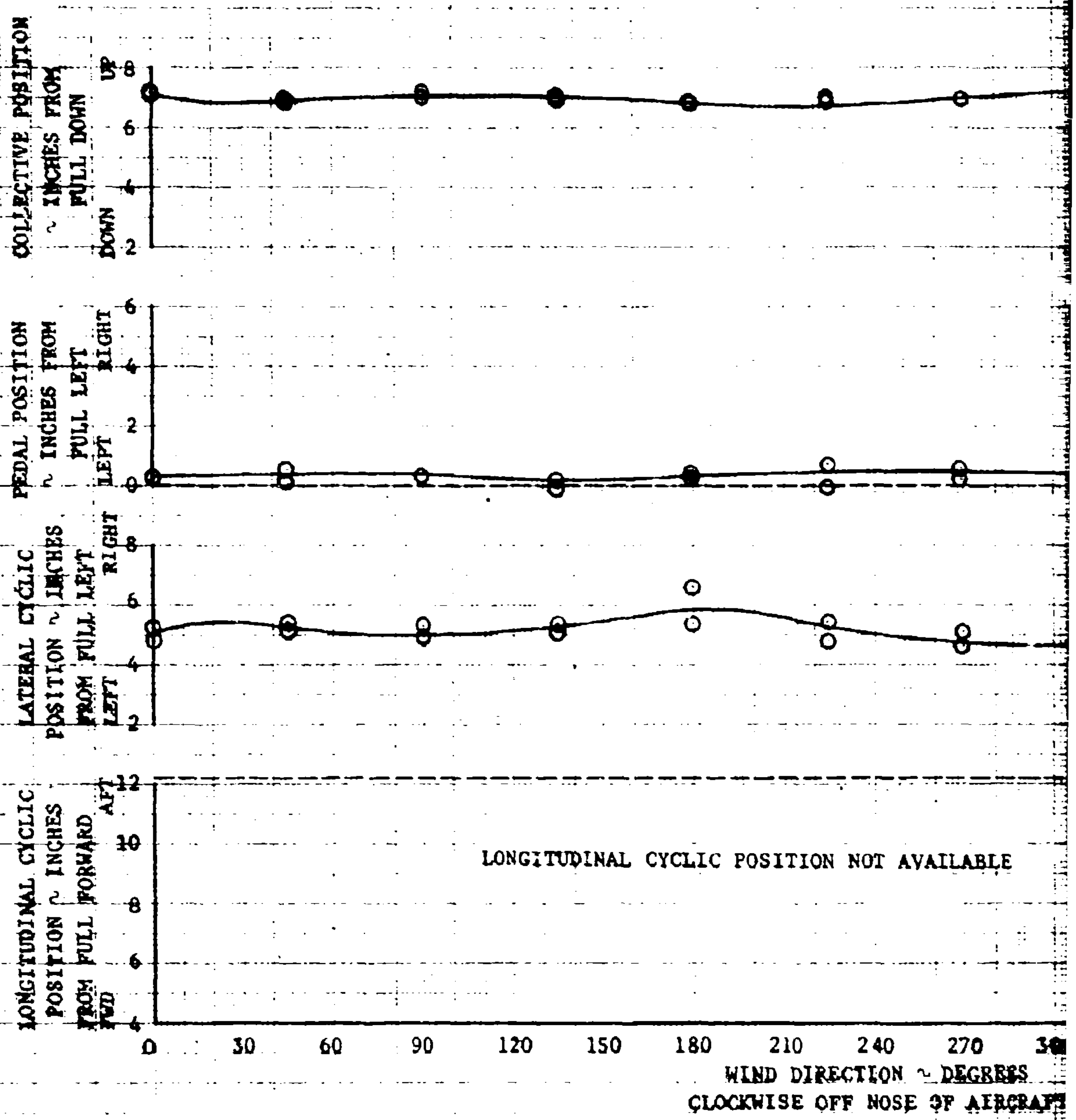
FIGURE 185

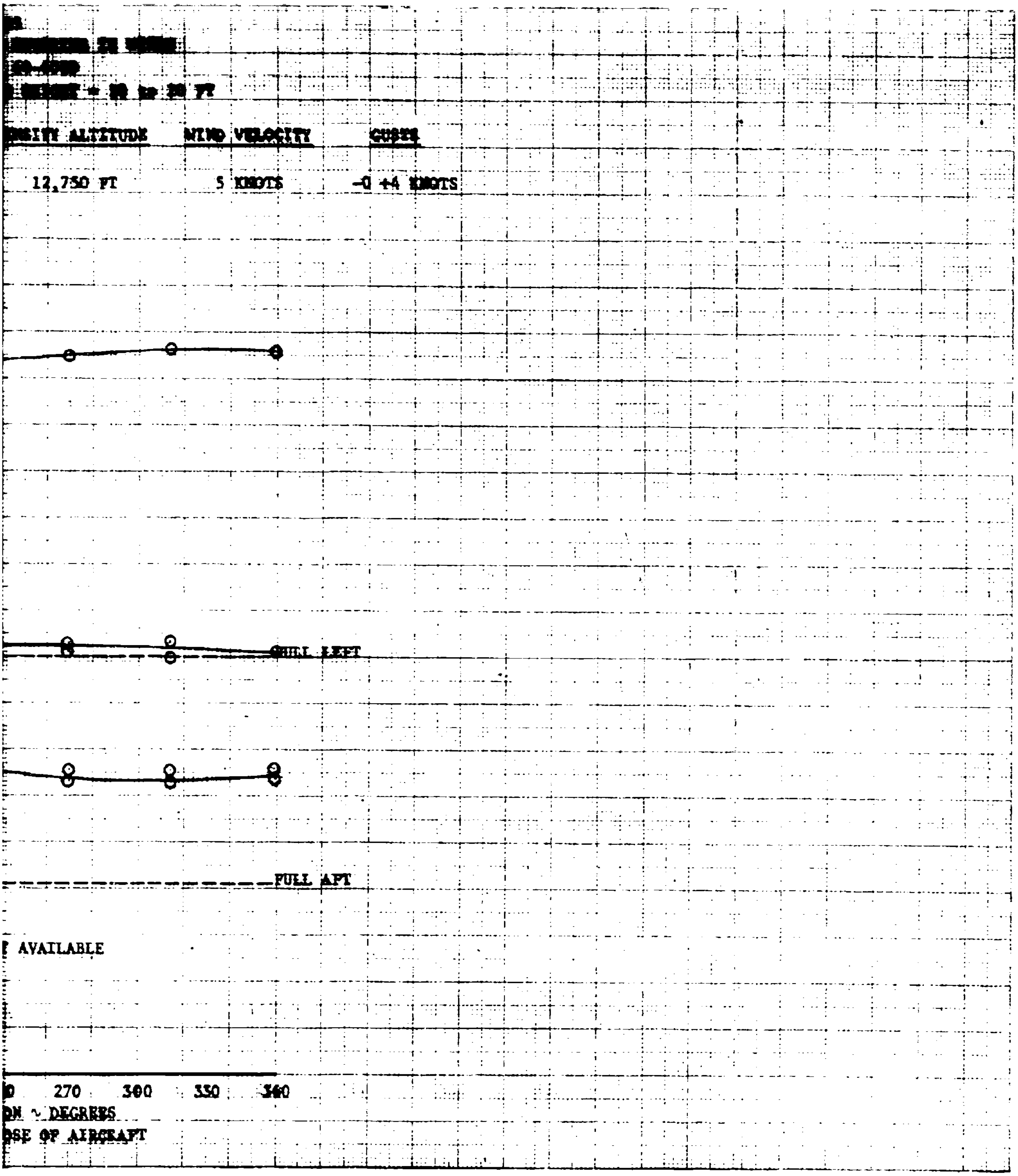
CONTROL POSITIONS WHILE HOVERING IN W

YUH-1H S/N 60-6029

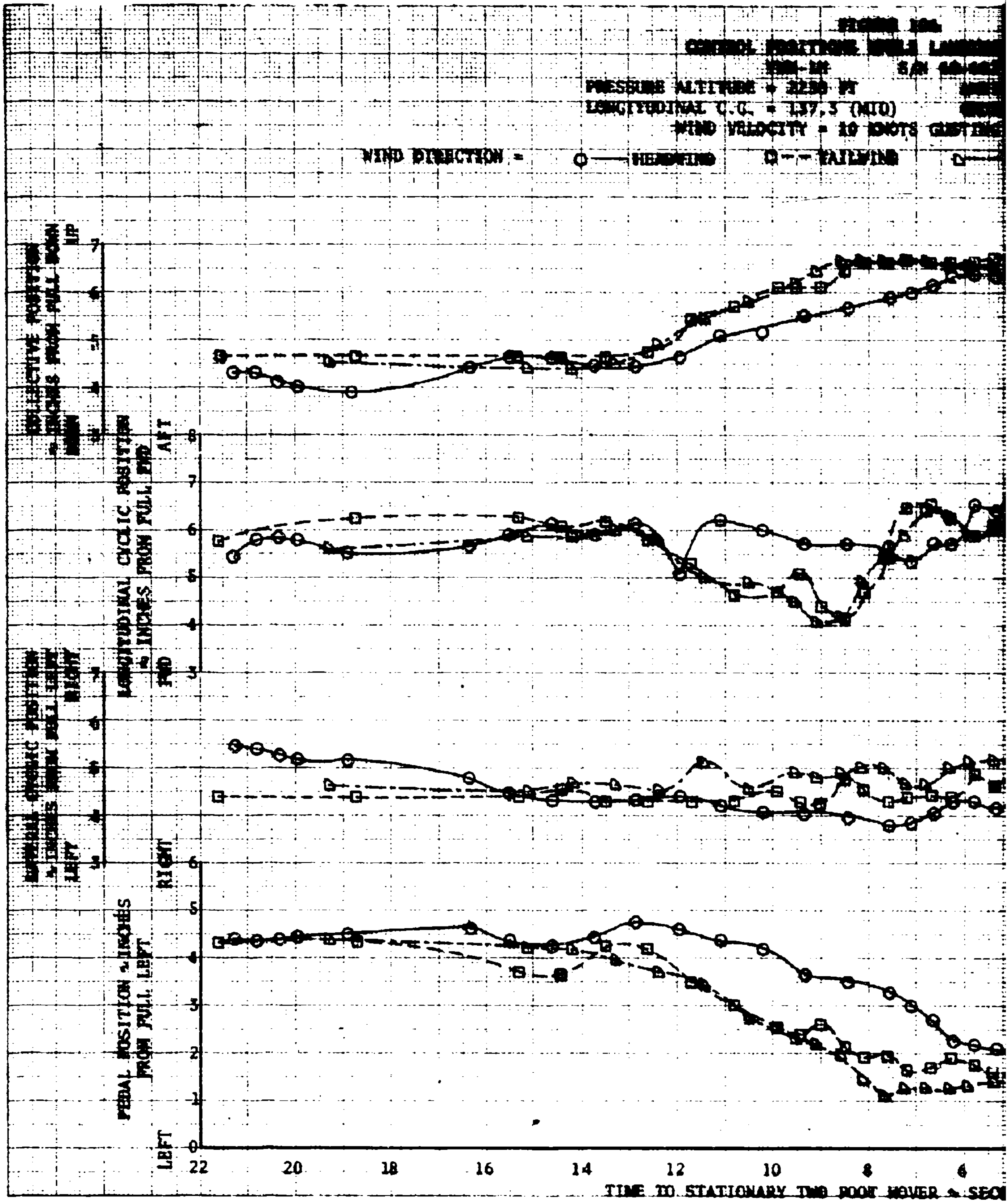
ROTOR SPEED = 324 RPM SKID HEIGHT = 10

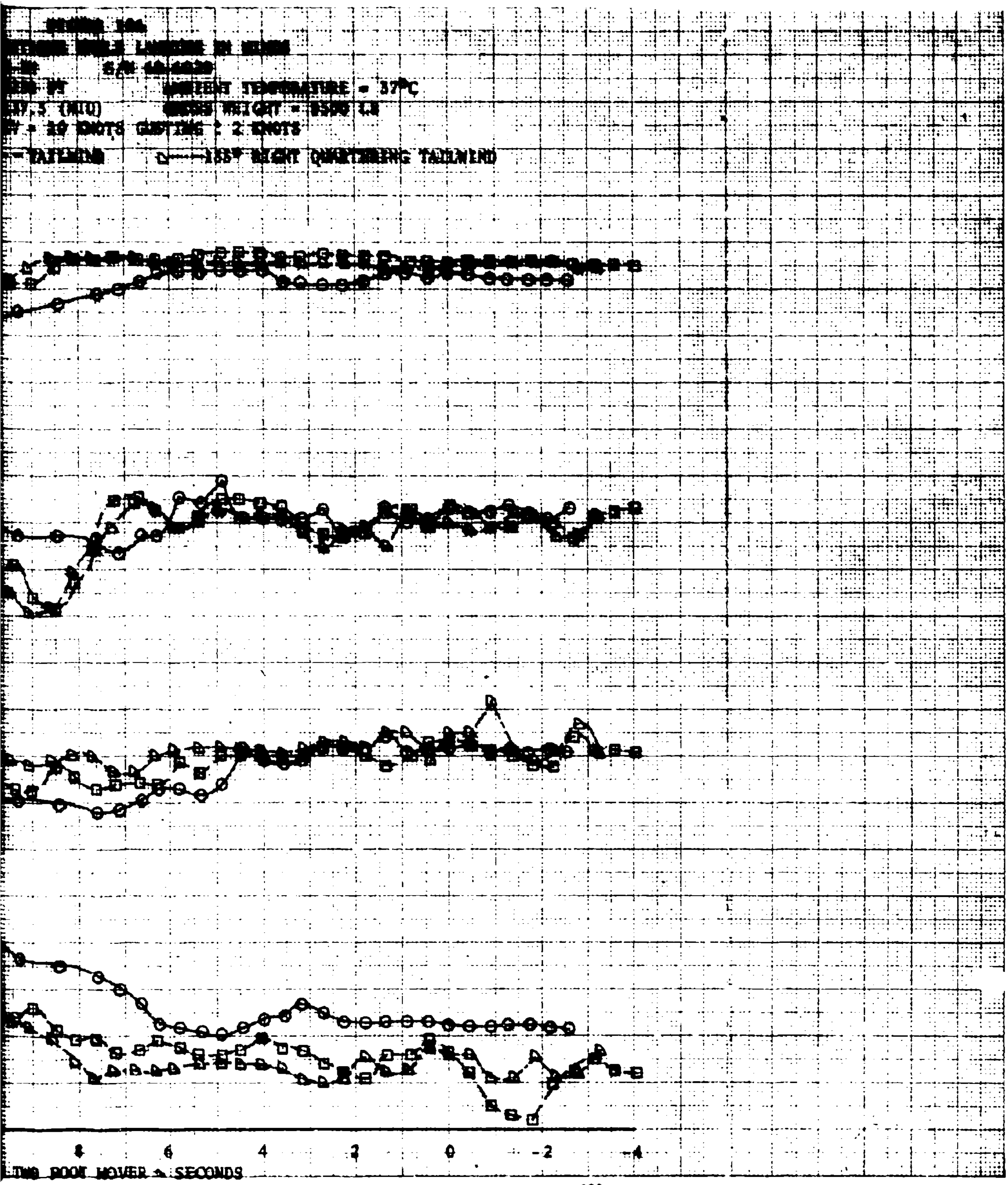
SYMBOL	WEIGHT	LONG CG	LAT CG	DENSITY ALTITUDE
O ———	8600 LB	134.0 (FWD)	0 (MID)	12,750 FT




ALTITUDE
VELOCITY
12,750 FT
5 KNOTS
-0 +4 KNOTS
FULL LEFT
FULL AFT
AVAILABLE
270
300
330
360
DEGREES
AIRCRAFT

2


PRESSURE ALTITUDE = 2230 FT
LONGITUDINAL C.G. = 137.3 (MID)
WIND VELOCITY = 10 KNOTS
WIND DIRECTION =
HEADWIND
TAILWIND
COLLECTIVE POSITION ~ INCHES FROM FULL DOWN
UP
LONGITUDINAL CYCLIC POSITION ~ INCHES FROM FULL FWD
AFT
FWD
RIGHT
LEFT
PEDAL POSITION ~ INCHES FROM FULL LEFT
RIGHT
LEFT
22
20
18
16
14
12
10
8
6
TIME TO STATIONARY TWO FOOT HOVER ~ SEC



AMBIENT TEMPERATURE = 37°C
KNOTS GUSTING ± 2 KNOTS
135° RIGHT QUARTERING TAILWIND
8
6
4
2
0
-2
-4
HOVER - SECONDS

2

FIGURE 107
POWER AND FLIGHT PATH WHILE
YUH-1H S/N 6

PRESSURE ALTITUDE = 2230 FT
LONGITUDINAL C.G. = 137.3 (MID)
WIND VELOCITY = 10 KNOTS GU

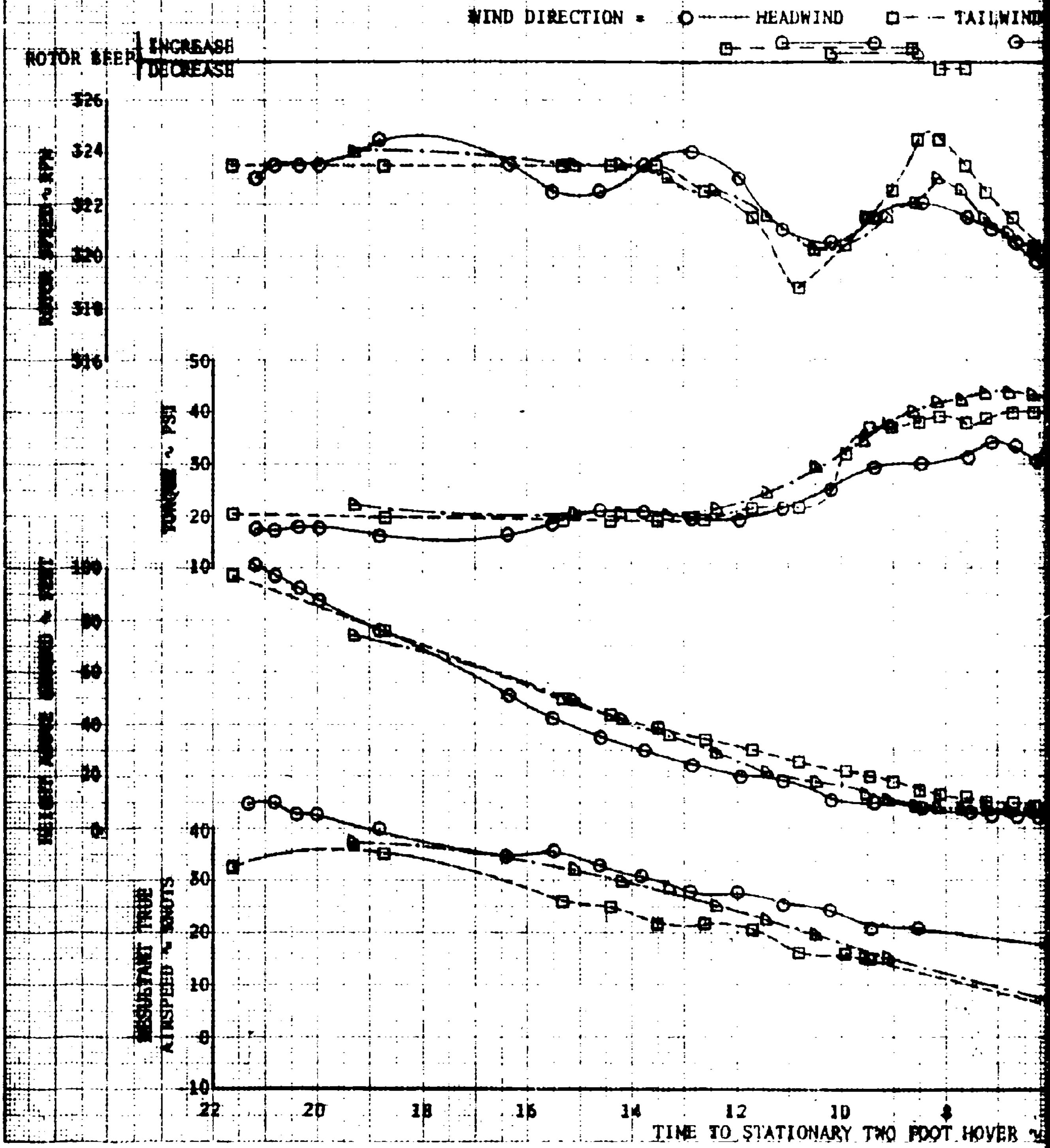
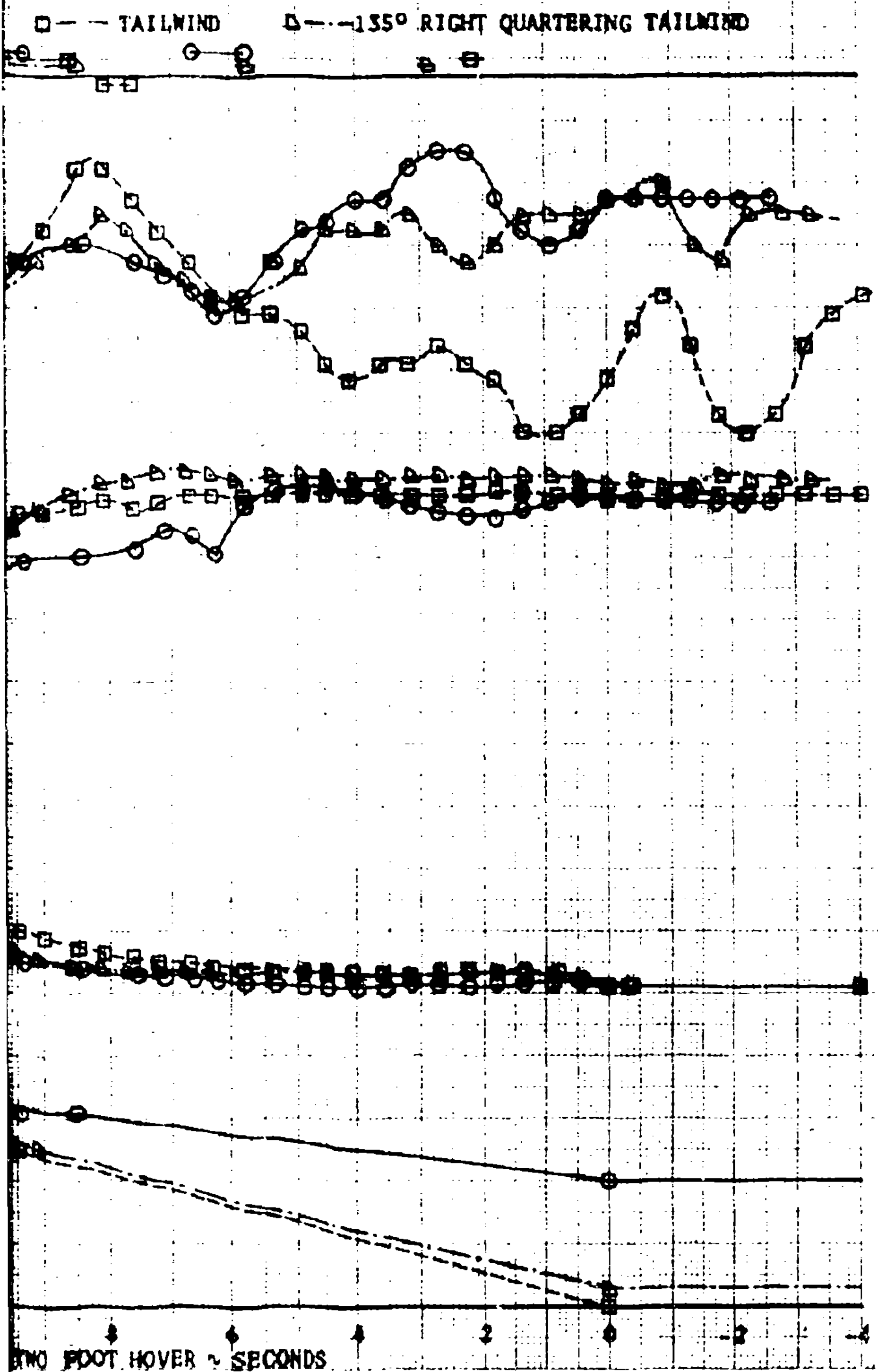


FIGURE 107
IGHT PATH WHILE LANDING IN WINDS
H-1H S/N 60-6029
2230 FT AMBIENT TEMPERATURE = 37°C
137.3 (MID) GROSS WEIGHT = 9500 LB
TY = 10 KNOTS GUSTING ± 2 KNOTS



2

FIGURE 108
ROTOR STATIC DROOP IN A HOVER
YUH-1B S/N 60-6029
T53-L-13 S/N LE 14001 AND LE 14008

O INCREASING COLLECTIVE □ DECREASING COLLECTIVE ● TRIM (RPM SET POINT)

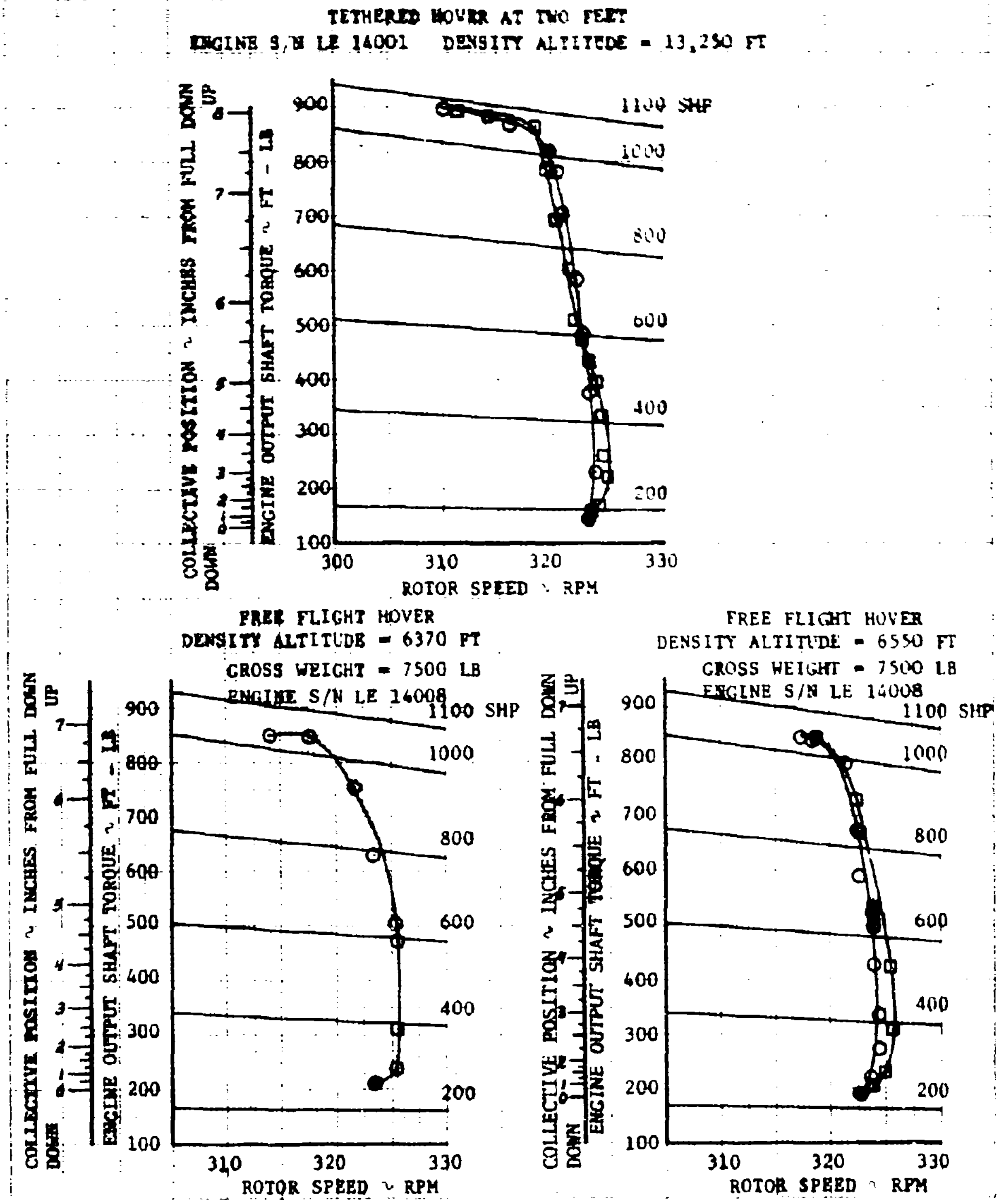


FIGURE 10[illegible]
ROTOR STATIC DROOP IN A CLIMB
YUH-[illegible] S/N 60-6029
T53-L-13 S/N LE 14008
AIRSPEED = 60 KCAS

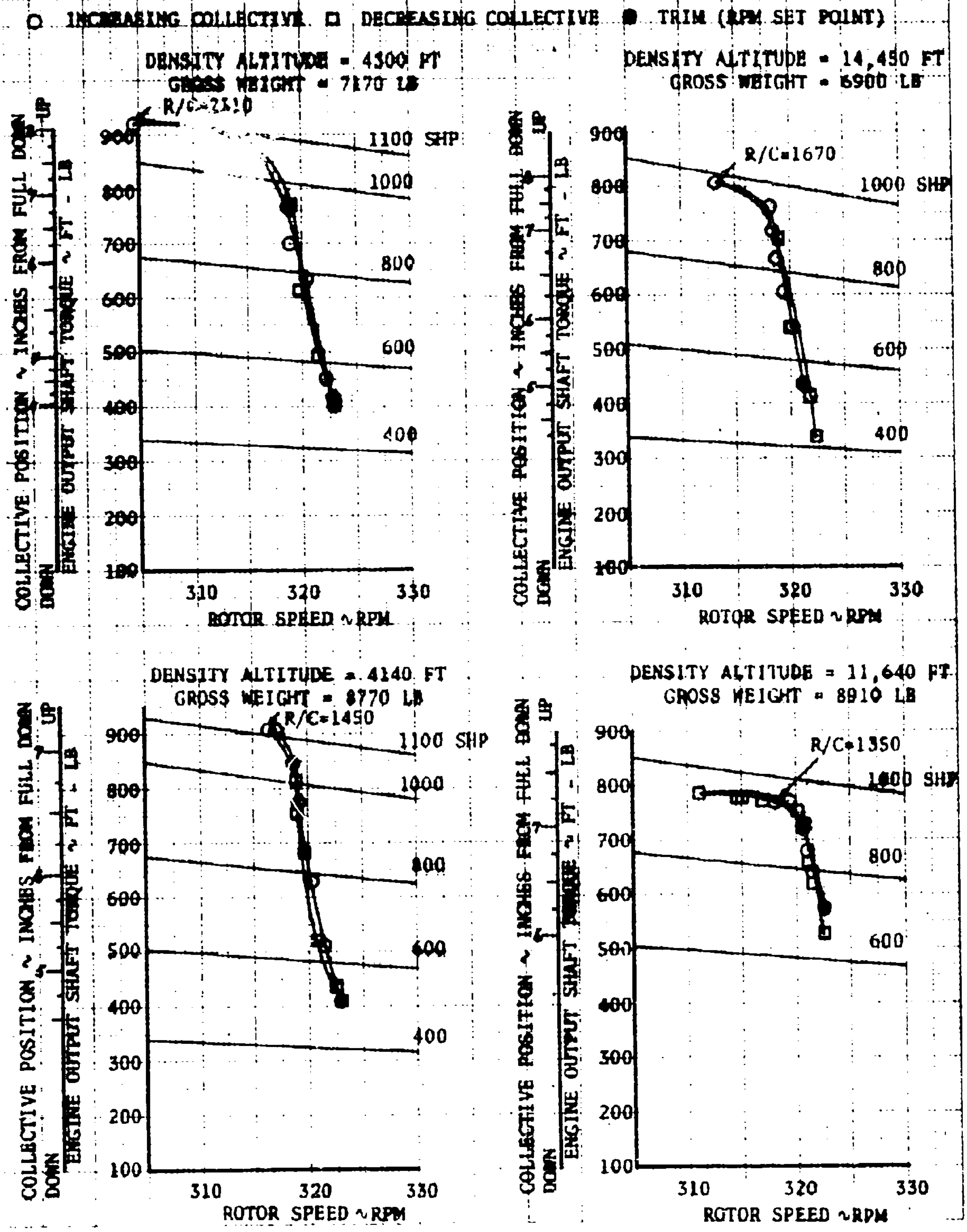


FIGURE 110
ROTOR [illegible] IN A DESCENT
[illegible] S/N [illegible]
T53-L-13 S/N LE 14008
AIRSPEED = 60 KCAS

○ INCREASING COLLECTIVE ⊡ DECREASING COLLECTIVE ● TRIM (RPM SET POINT)

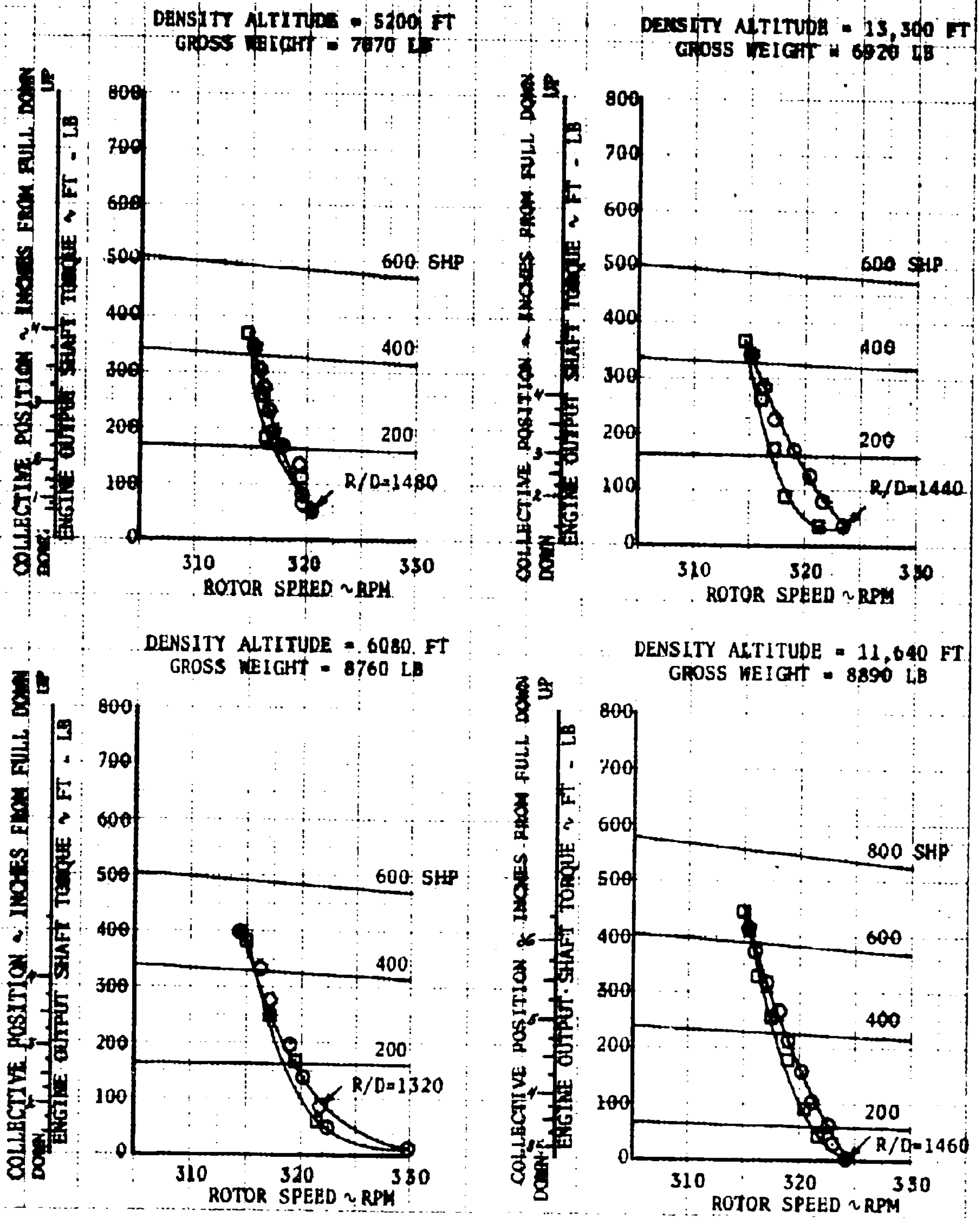


FIGURE 11A
ROTOR STATIC DROOP AT HIGH AIRSPEED
YUH-1H S/N 60-5029
T53-L-13 S/N LE 14000
LEVEL FLIGHT

○ INCREASING COLLECTIVE □ DECREASING COLLECTIVE ● TRIM (SET RPM POINT)

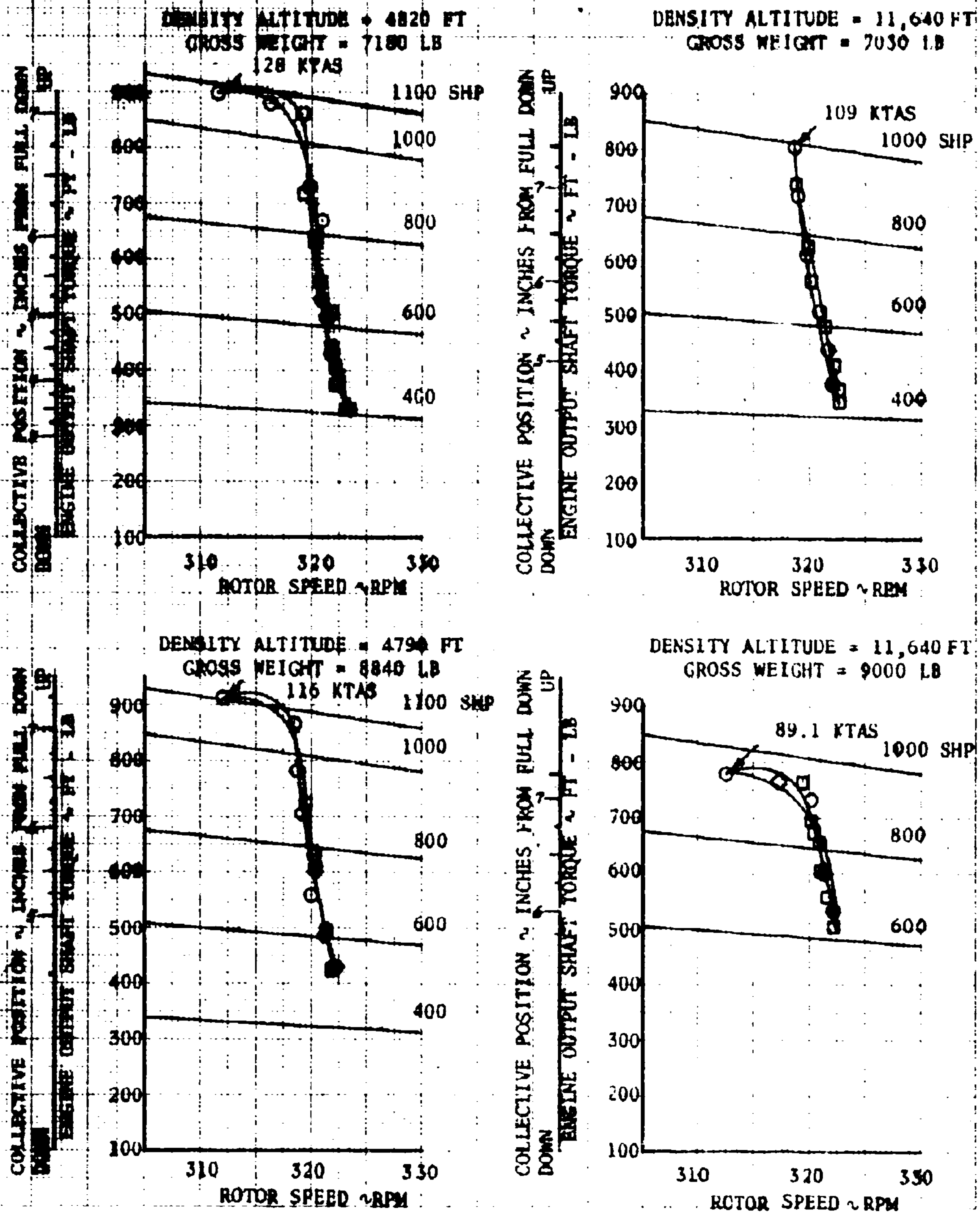
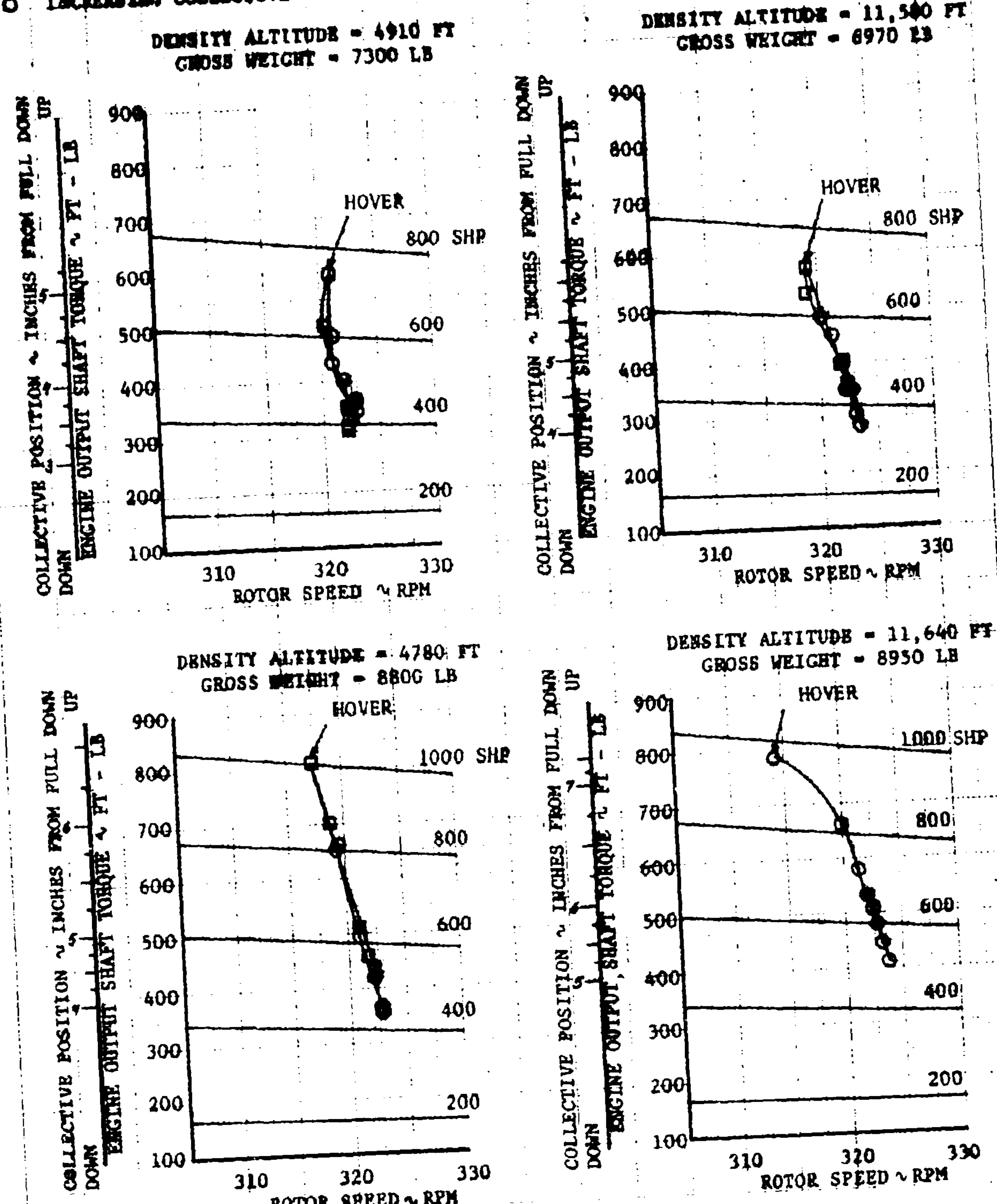
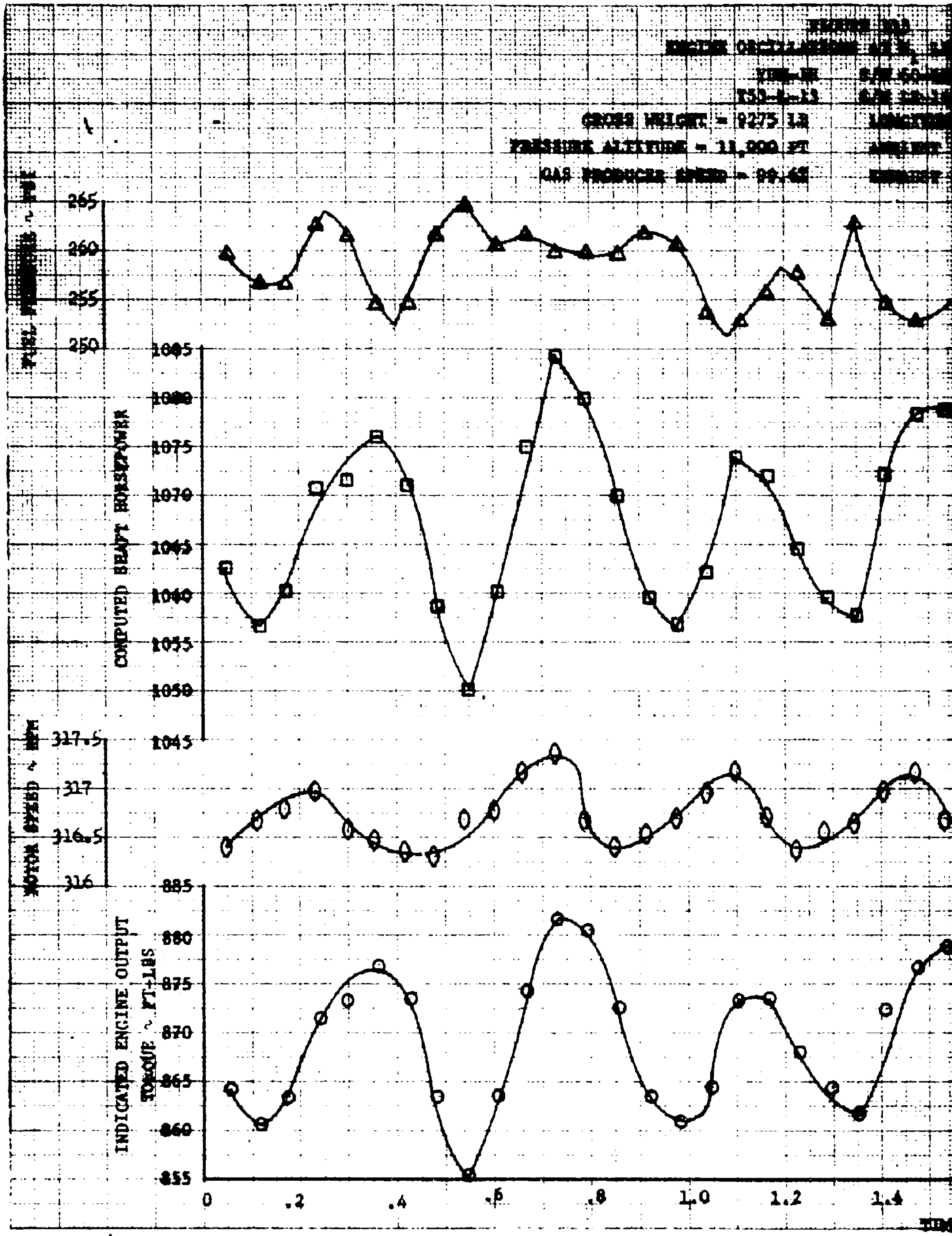


FIGURE 112
ROTOR STATIC DROOP AT LOW SPEED
YUH-1H S/N 60-6028
T53-L-13 S/N LE 14068
LEVEL FLIGHT

○ INCREASING COLLECTIVE □ DECREASING COLLECTIVE ● TRIM (RPM SET POINT)




GROSS WEIGHT = 9275 LB
COMPUTED SHAFT HORSEPOWER
INDICATED ENGINE OUTPUT TORQUE ~ FT-LBS
265
260
255
250
1085
1080
1075
1070
1065
1060
1055
1050
1045
317.5
317
316.5
316
885
880
875
870
865
860
855
0
.2
.4
.6
.8
1.0
1.2
1.4

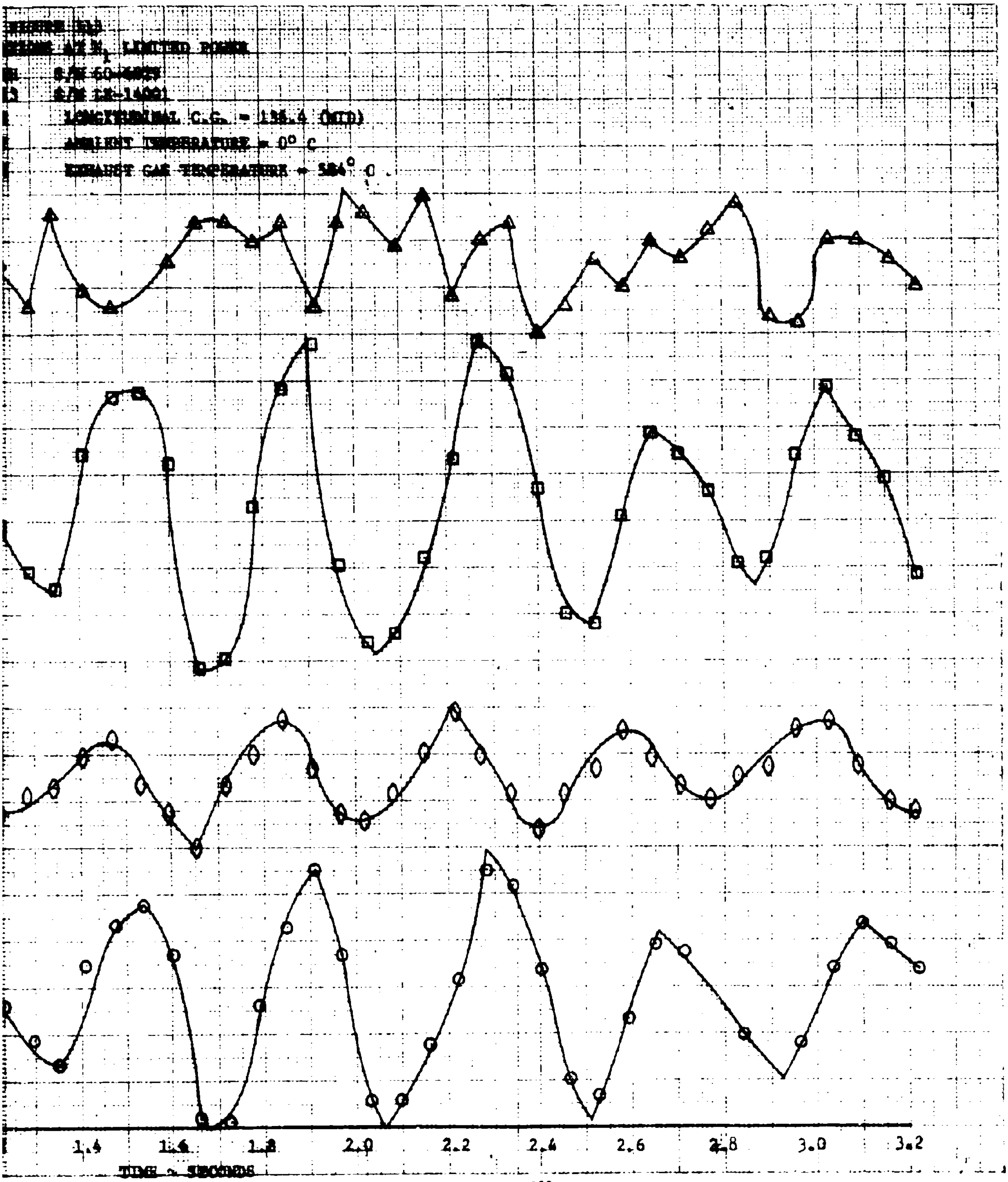

AMBIENT TEMPERATURE = 0° C
1.4
1.6
1.8
2.0
2.2
2.4
2.6
2.8
3.0
3.2
TIME ~ SECONDS

FIGURE 114
ENGINE INLET CHARACTERISTICS

YUH-1H T53-L-13
S/N 60-6029 S/N LE 14001

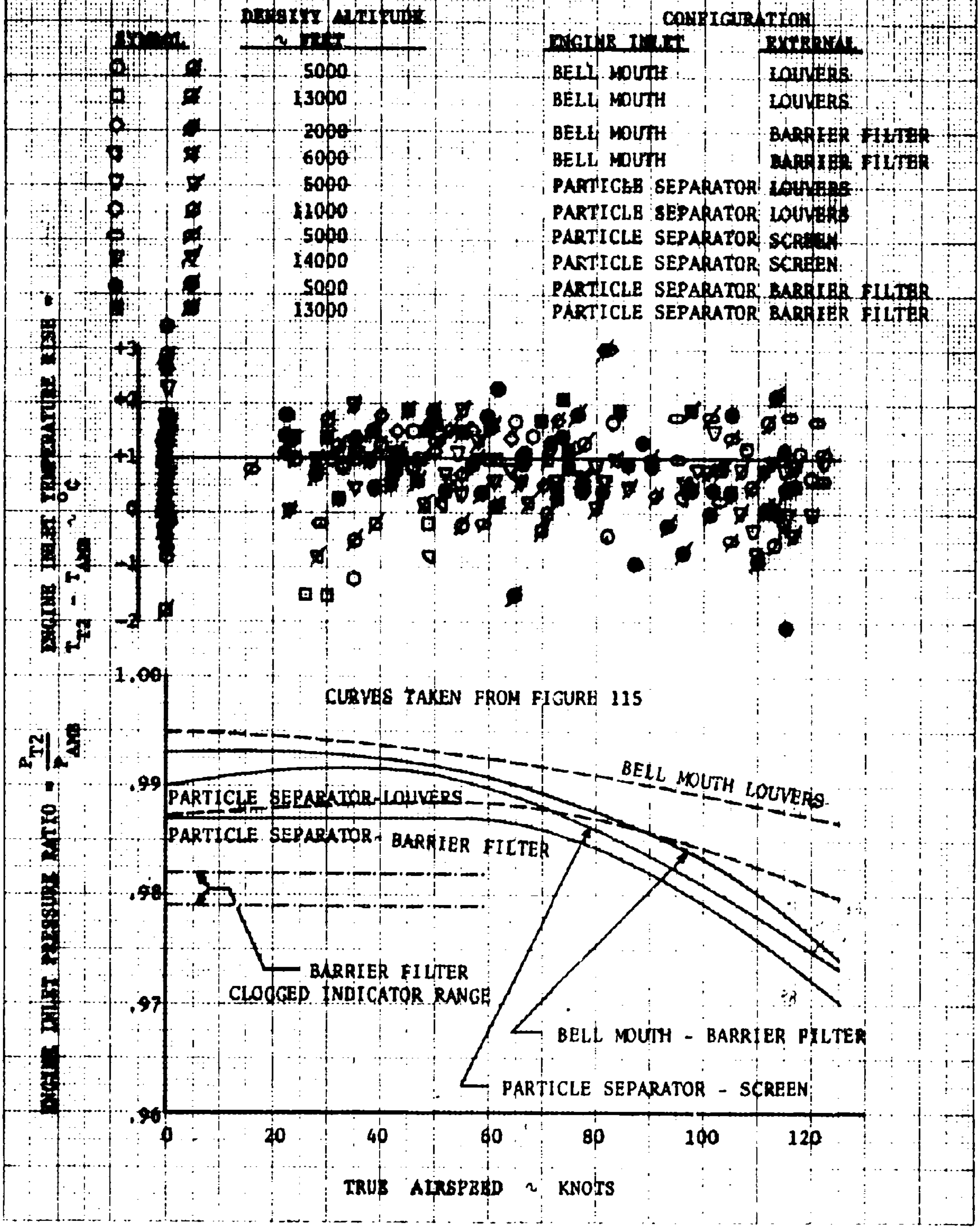


FIGURE 115
ENGINE INLET PRESSURE RATIOS
A/C YUH-1H   ENGINE T53-L-13
S/N 60-6029   S/N LE 14001

SYMBOL	DENSITY ALTITUDE	CONFIGURATION ENGINE INLET	CONFIGURATION EXTERNAL
○	5000	BELL MOUTH	LOUVERS
□	13000	BELL MOUTH	LOUVERS
◇	2000	BELL MOUTH	BARRIER FILTER
◁	6000	BELL MOUTH	BARRIER FILTER
▽	5000	PARTICLE SEPARATOR	LOUVERS
○	11000	PARTICLE SEPARATOR	LOUVERS
○	5000	PARTICLE SEPARATOR	SCREEN
▽	14000	PARTICLE SEPARATOR	SCREEN
●	5000	PARTICLE SEPARATOR	BARRIER FILTER
■	13000	PARTICLE SEPARATOR	BARRIER FILTER

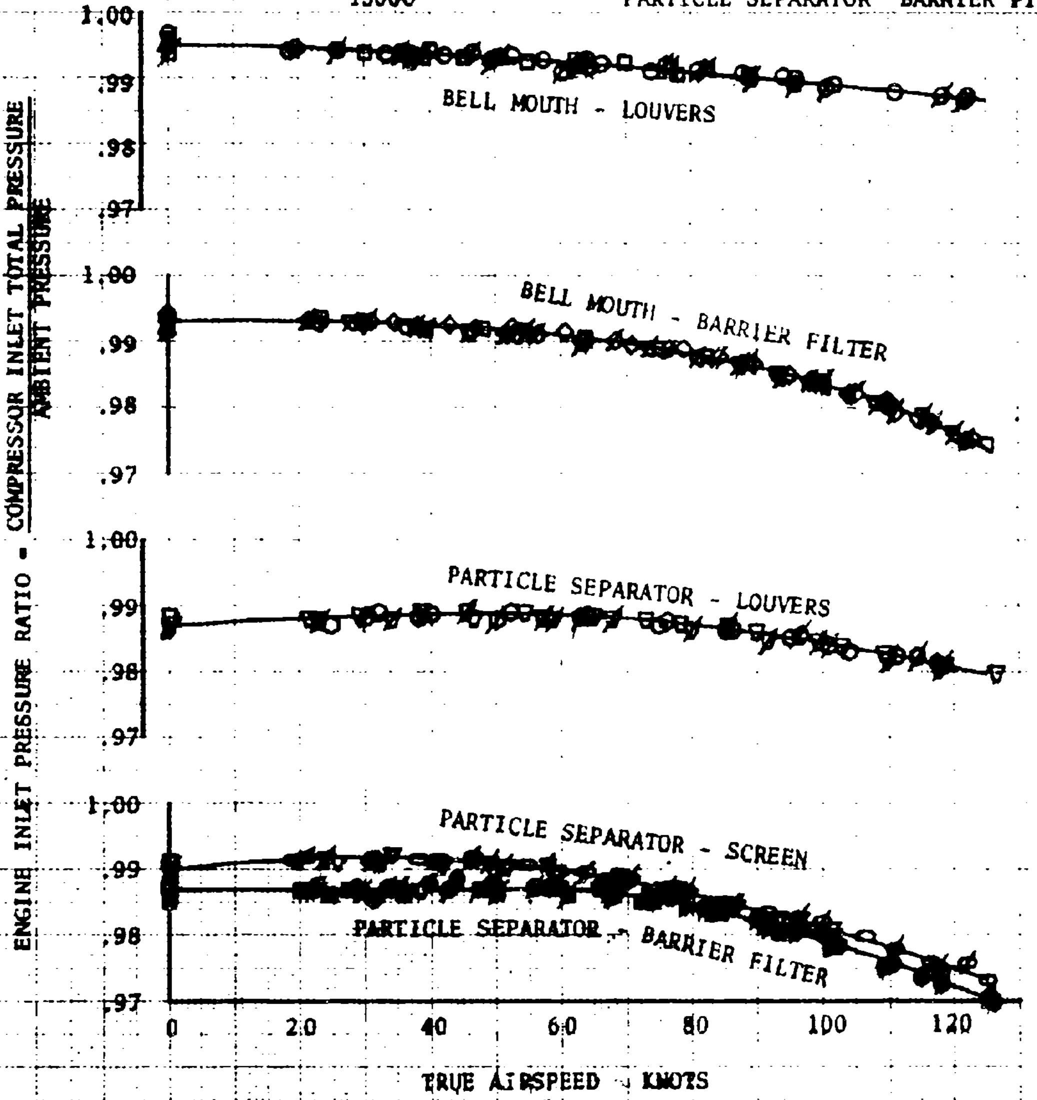


FIGURE 116

MILITARY (30-MINUTE LIMIT) SHAFT HORSEPOWER AVAILABLE

UH-1H T53-L-13

6600 ENGINE OUTPUT SHAFT RPM

NOTE: Based on Model LTC1K-4 Specification on No. 104.33 corrected for the following installation conditions:

1. Compressor inlet temperature rise = 1°C
2. Compressor inlet pressure ratio = 0.987
3. Air bleed = 1.15% of total air flow
4. Horsepower extracted = 0.4 Horsepower
5. Exhaust losses = Zero
6. Bleed air heater OFF
7. Anti-ice OFF

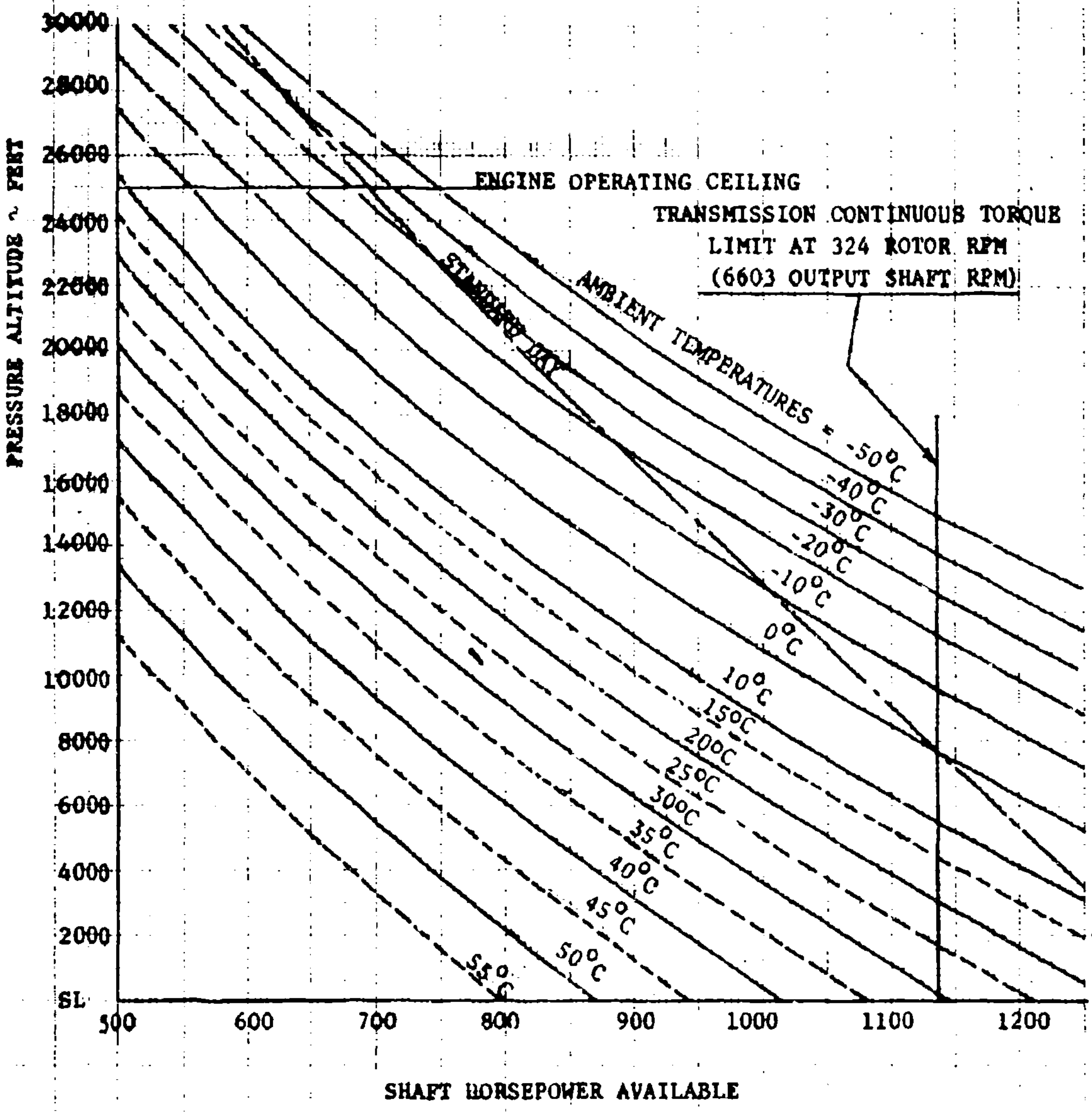


FIGURE 117

MAXIMUM CONTINUOUS SHAFT HORSEPOWER AVAILABLE

UH-1H T53-L-13

6600 ENGINE OUTPUT SHAFT RPM

NOTE: Based on Model LTC1K-4 Specification No. 104.33 corrected for the following installation conditions:

1. Compressor inlet temperature rise = 1°C
2. Compressor inlet pressure ratio = 0.987
3. Air Bleed = 1.15% of total air flow
4. Horsepower extracted = 0.4 Horsepower
5. Exhaust losses = Zero
6. Bleed air heater OFF
7. Anti-ice OFF

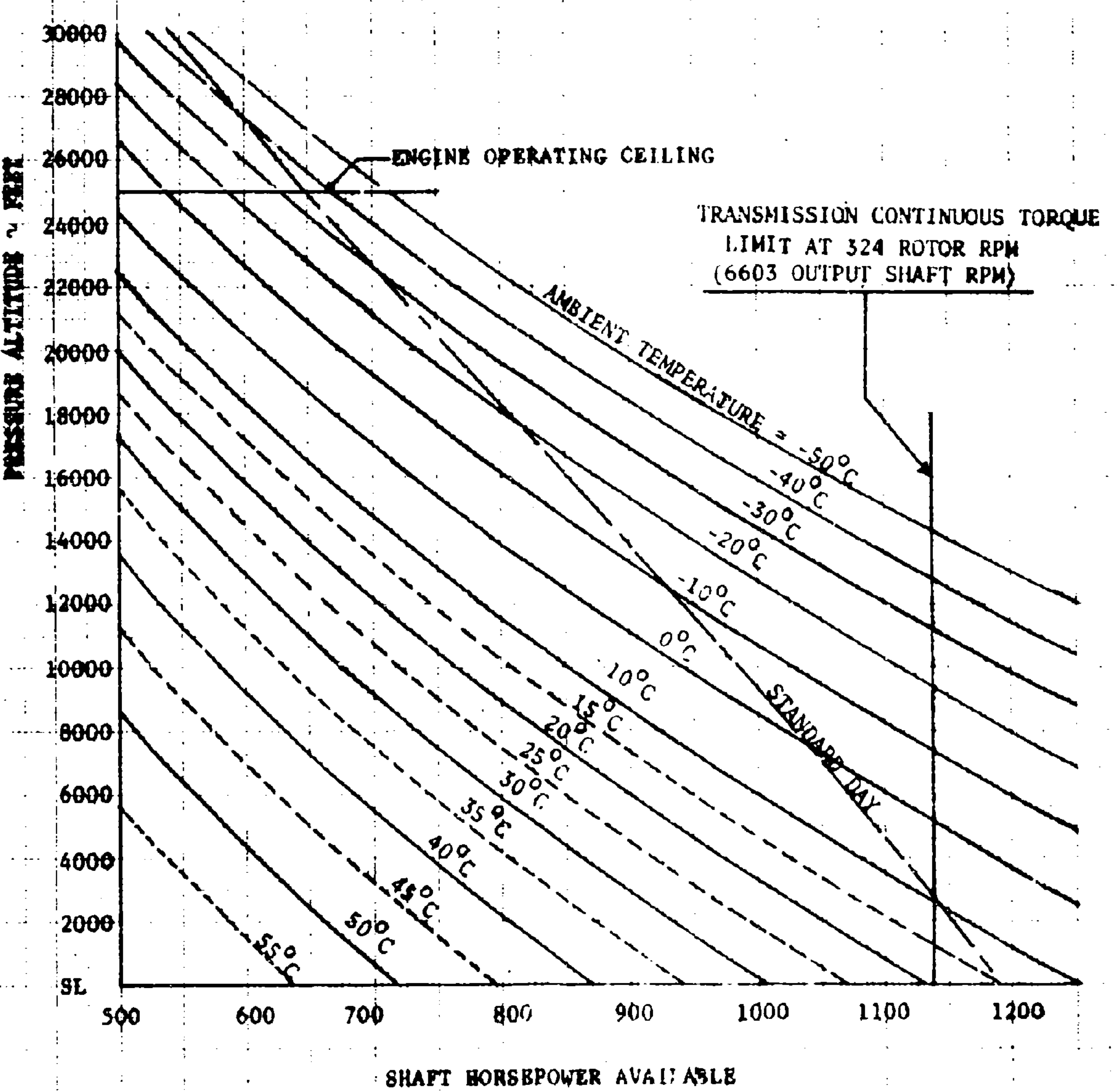


FIGURE 118

MAXIMUM CONTINUOUS SHAFT HORSEPOWER AVAILABLE

UH-1H T53-L-13

6400 ENGINE OUTPUT SHAFT RPM

NOTE: Based on Model LTC1K-4 Specification No. 104.33 corrected for the following installation conditions:

1. Compressor inlet temperature rise = 1°C
2. Compressor inlet pressure ratio = 0.987
3. Air bleed = 1.15% of total air flow
4. Horsepower extracted = 0.4 Horsepower
5. Exhaust losses = Zero
6. Bleed air heater = OFF
7. Anti-ice OFF

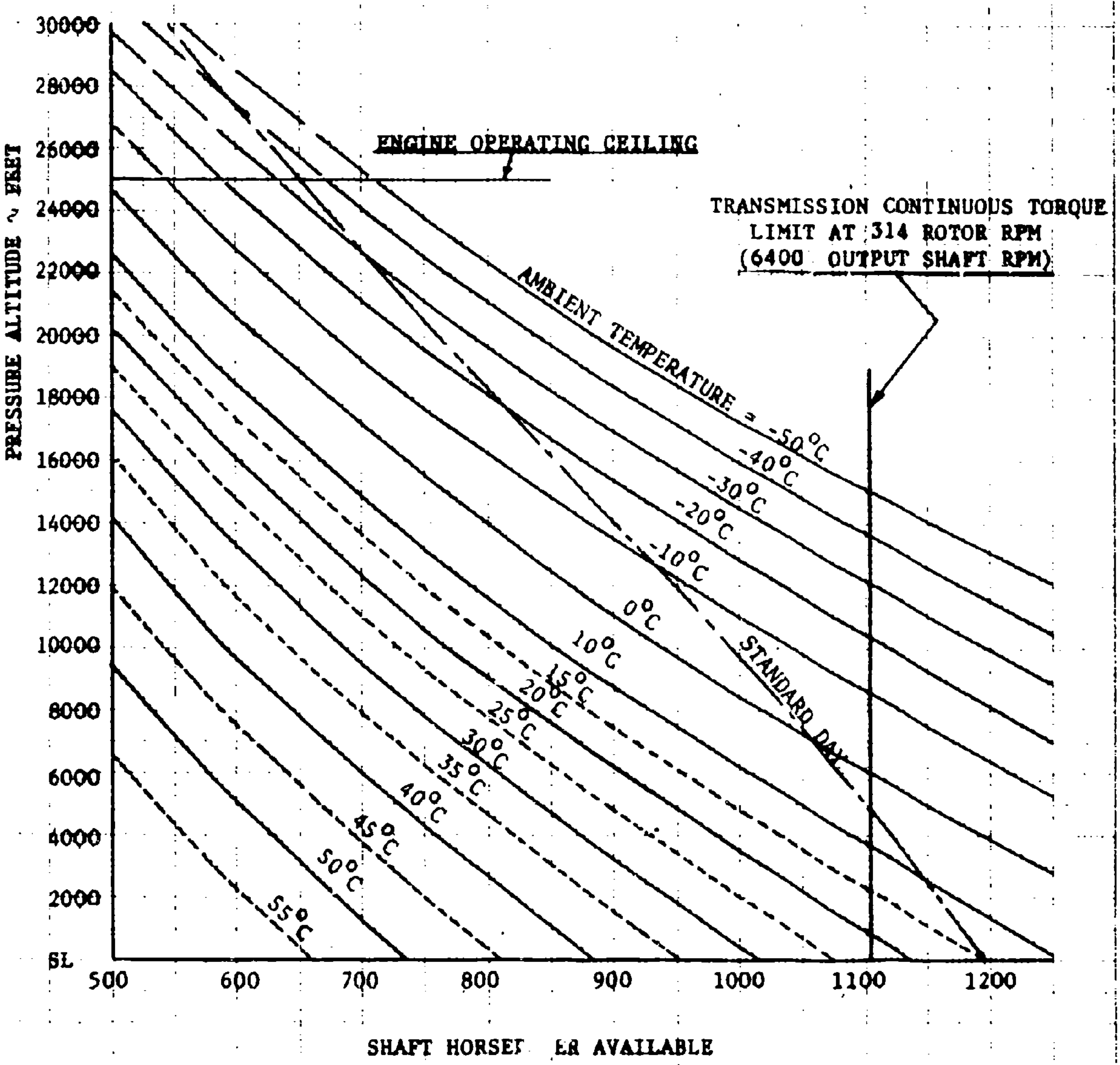


FIGURE 119
EFFECT OF AIRSPEED ON
MAXIMUM CONTINUOUS POWER AVAILABLE
UH-1H T53-L-13
6600 OUTPUT SHAFT RPM
STANDARD DAY

NOTE: Based on Model LTC1K-4 Specification No. 104.33
Corrected for the following installation conditions:

1. Compressor inlet temperature rise = 1°C
2. Air bleed = 1.15% of total airflow
3. Horsepower extracted = 0.4 Horsepower
4. Exhaust losses = Zero
5. Bleed air heater = OFF
6. Anti-Ice = OFF
7. Compressor inlet pressure ratios shown for particle separator - barrier filter on figure 115

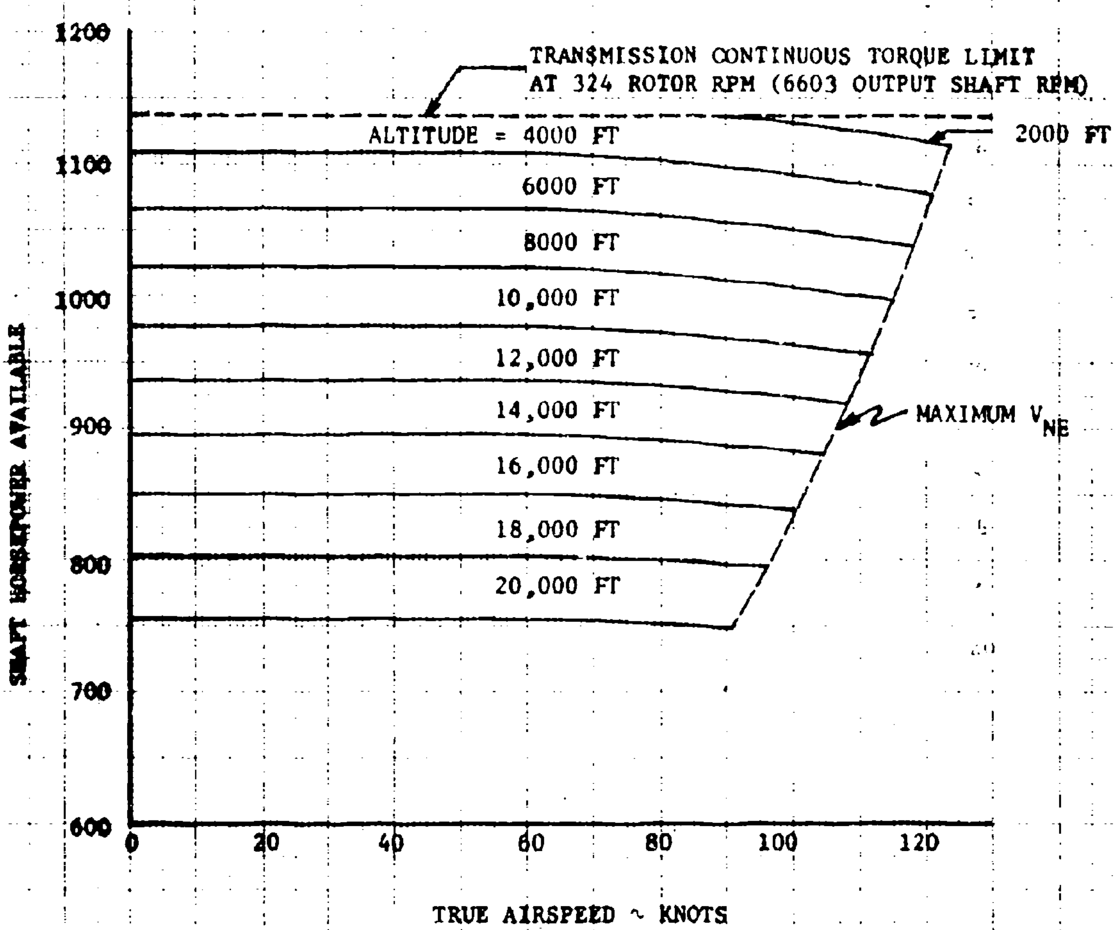


FIGURE 120
MAXIMUM CONTINUOUS SHAFT HORSEPOWER AVAILABLE
UH-1H T53-L-13
6600 ENGINE OUTPUT SHAFT RPM
ANTI-ICE ON BLEED AIR HEAT FULL ON

NOTE: Based on Model LTC1K-4 Specification No. 104.33 corrected for the following installation conditions:
1. Compressor inlet temperature rise = 1°C
2. Compressor inlet pressure ratio = 0.987
3. Air Bleed = 3.5% of total air flow
4. Horsepower extracted = 0.4 Horsepower
5. Exhaust losses = Zero

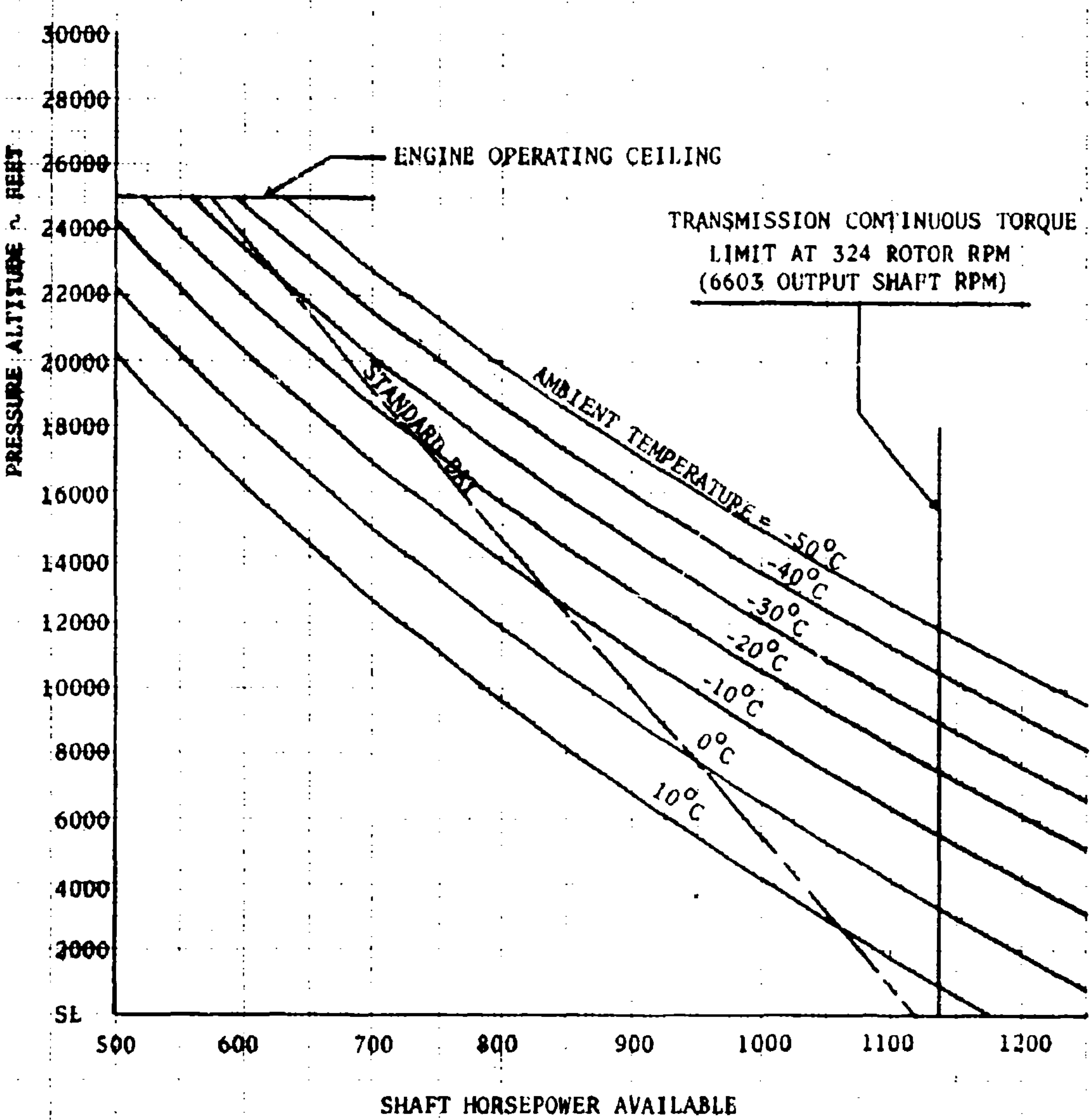


FIGURE 121

SPECIFICATION FUEL FLOW

UH-1H T53-L-13

STANDARD DAY 6600 OUTPUT SHAFT RPM

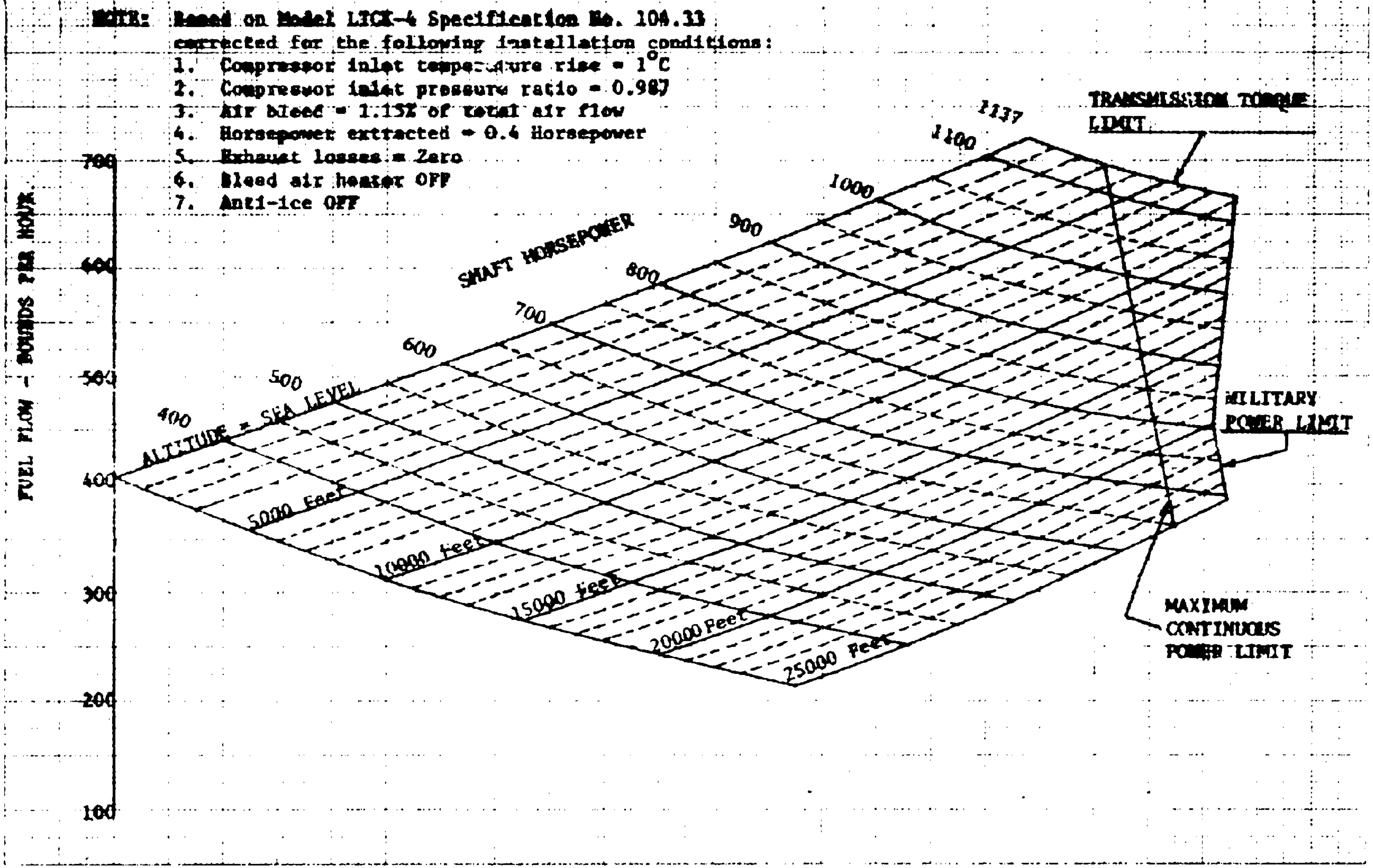


FIGURE 122

SPECIFICATION FUEL FLOW

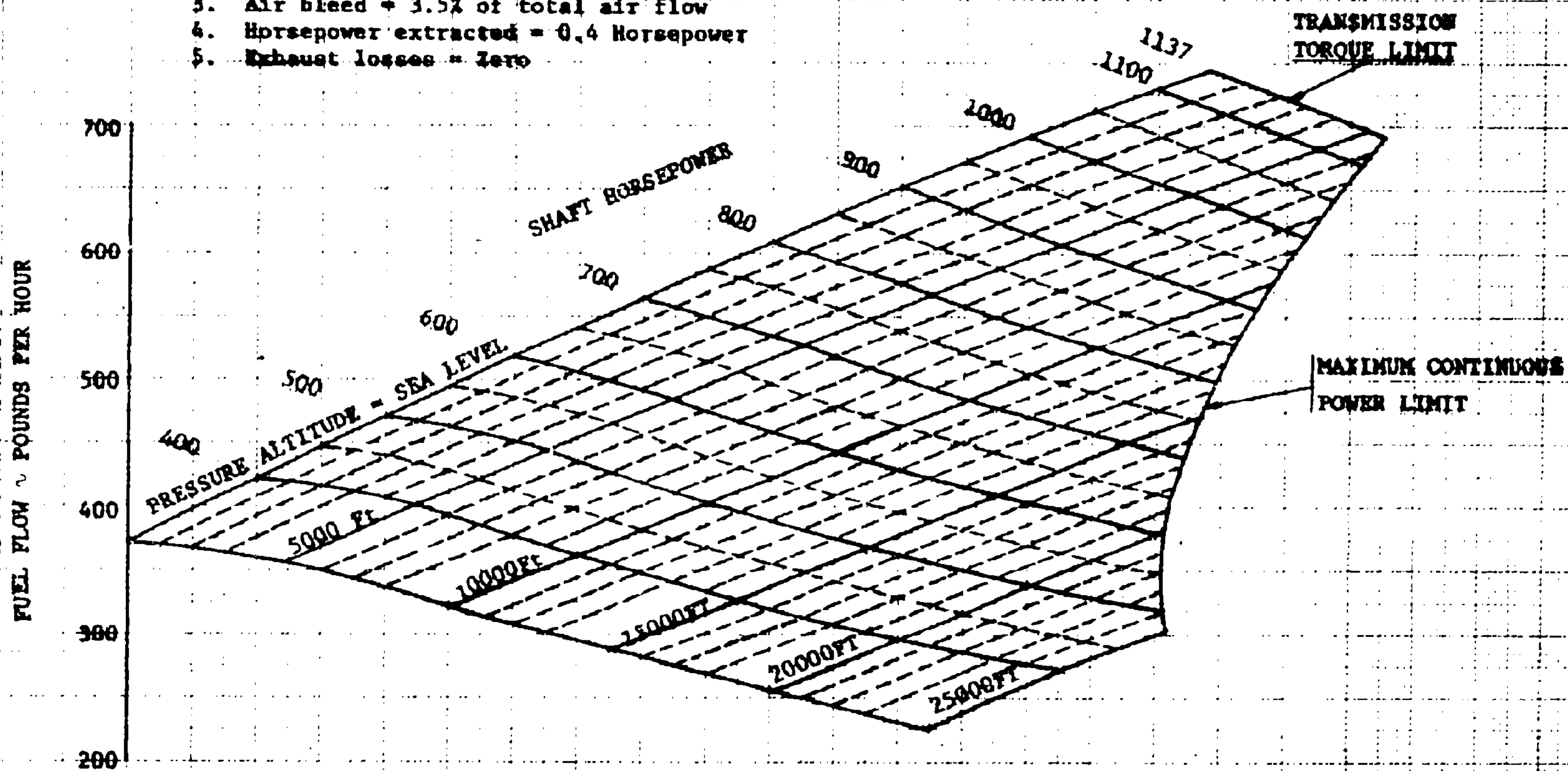
UH-2H T53-L-13

AMBIENT TEMPERATURE = -10°C 6600 OUTPUT SHAFT RPM

ANTI-ICE ON BLEED AIR HEAT FULL ON

NOTE: Based on Model LTCK-4 Specification No. 104.33 corrected for the following installation conditions:

1. Compressor inlet temperature rise = 1°C
2. Compressor inlet pressure ratio = 0.987
3. Air bleed = 3.5% of total air flow
4. Horsepower extracted = 0.4 Horsepower
5. Exhaust losses = Zero



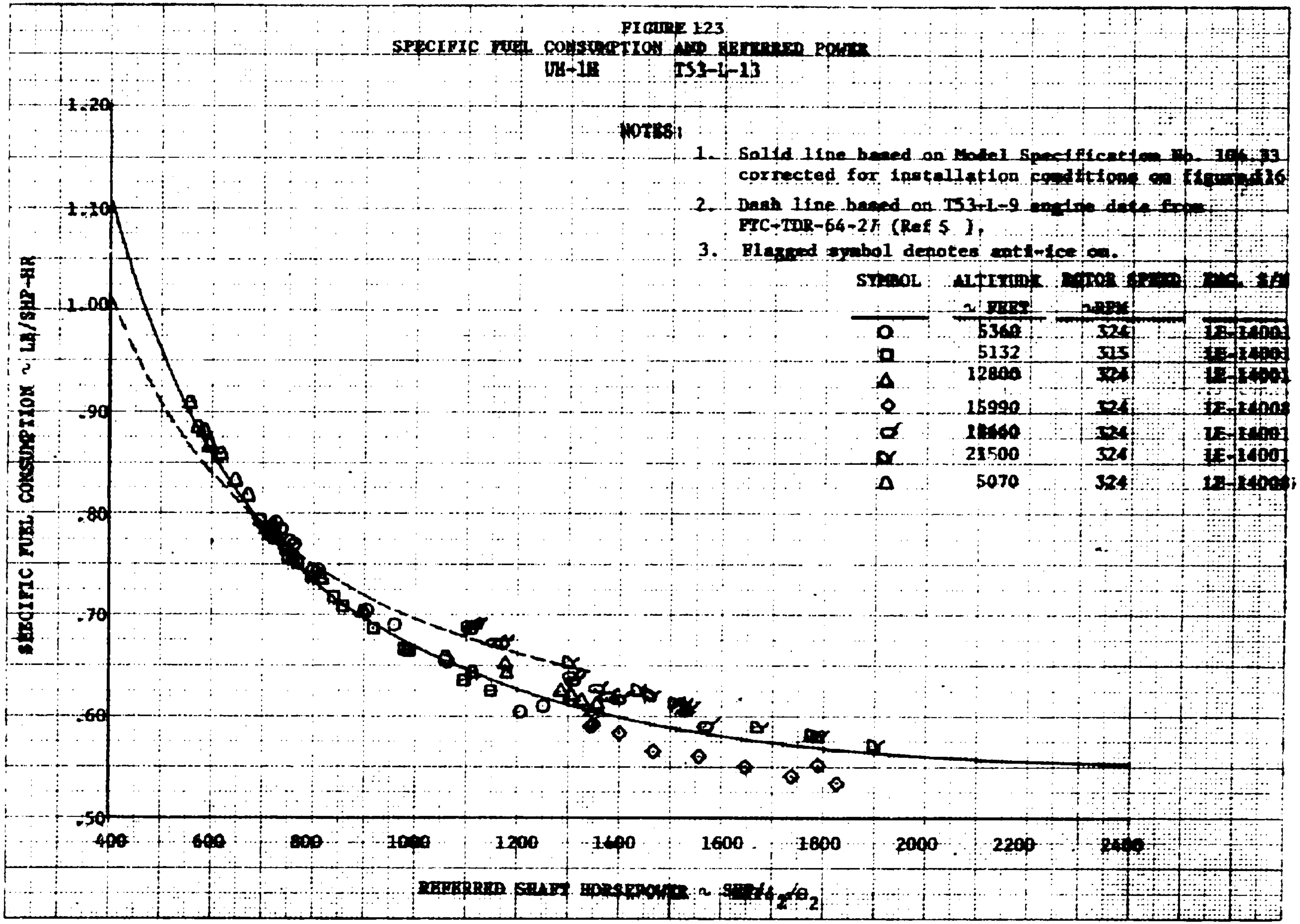


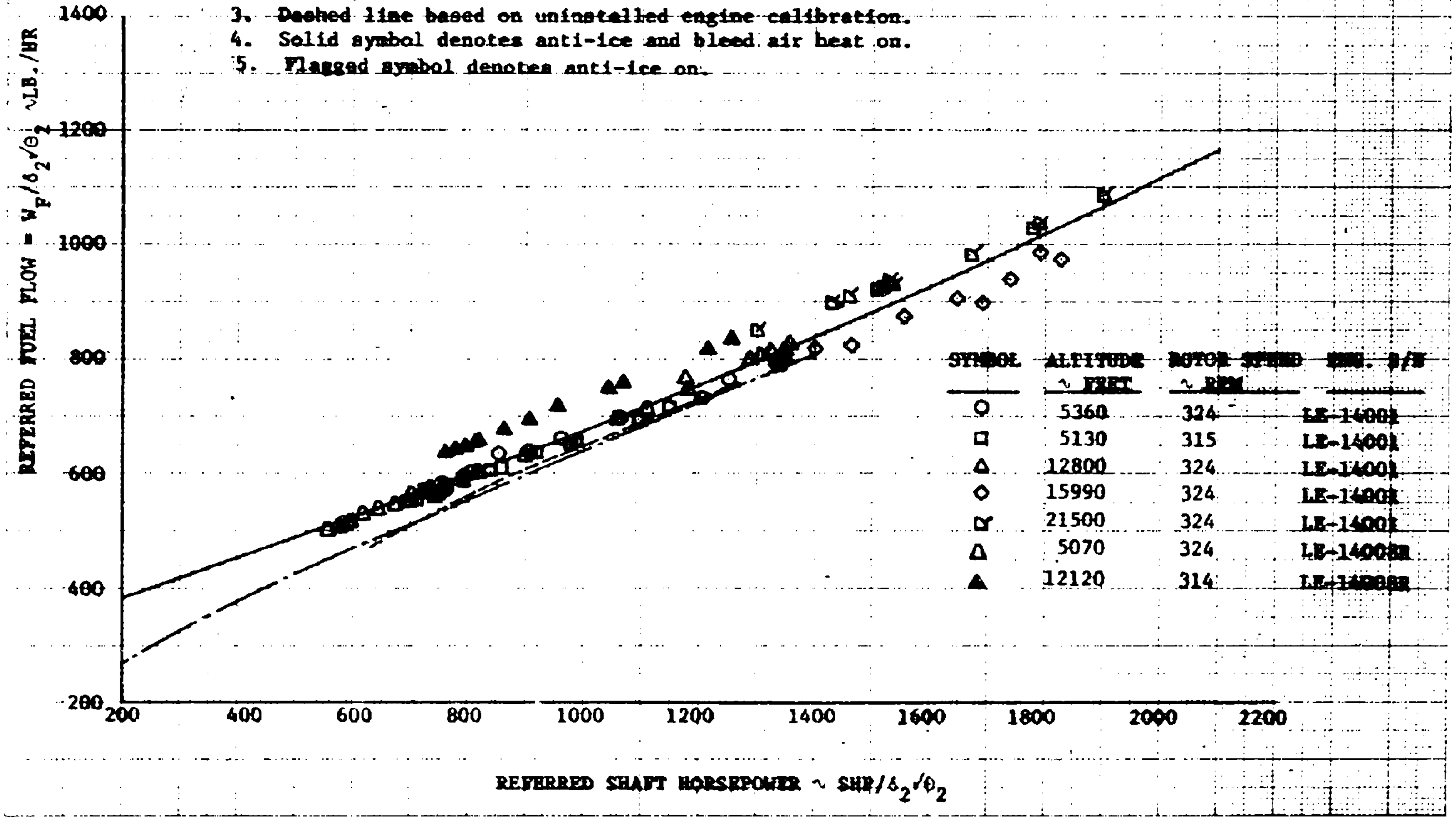
FIGURE 123
SPECIFIC FUEL CONSUMPTION AND REFERRED POWER
UH-1H T53-L-13
NOTES:
1. Solid line based on Model Specification No. 104.33 corrected for installation conditions on figure 16
2. Dash line based on T53-L-9 engine data from FTC-TDR-64-27 (Ref 5),
3. Flagged symbol denotes anti-ice on.
SYMBOL
ALTITUDE ~ FEET
ROTOR SPEED ~ RPM
5360 324
5132 315
12800 324
15990 324
21500 324
5070 324
SPECIFIC FUEL CONSUMPTION ~ LB/SHP-HR
1.20
1.10
1.00
.90
.80
.70
.60
.50
400
600
800
1000
1200
1400
1600
1800
2000
2200
2400
REFERRED SHAFT HORSEPOWER

FIGURE 124
REFERRED FUEL FLOW AND POWER

UH-1H T53-L-13

NOTES:

1. Solid line based on Model Specification No. 104.33 corrected for installation conditions on figure. 116
2. Dot-dashed line based on Model Specification No. 104.33 uninstalled.
3. Dashed line based on uninstalled engine calibration.
4. Solid symbol denotes anti-ice and bleed air heat on.
5. Flagged symbol denotes anti-ice on.



SYMBOL	ALTITUDE ~ FEET	ROTOR SPEED ~ RPM	ENG. S/N
○	5360	324	LE-14001
□	5130	315	LE-14001
△	12800	324	LE-14001
◇	15990	324	LE-14001
⊏ (flagged)	21500	324	LE-14001
△	5070	324	LE-14008R
▲	12120	314	LE-14008R

FIGURE 125

REFERRED GAS PRODUCER SPEED AND POWER

UH-1H T53-L-13

NOTES:

1. 100% N_1 = 25,150 RPM
2. Solid line based on Model Specification No. 104.33 corrected for installation conditions on figure 116
3. Dot-dashed line based on Model Specification No. 104.33 uninstalled.
4. Dashed line based on uninstalled engine calibration.
5. Solid symbol denotes anti-ice and bleed air heat on.
6. Flagged symbol denotes anti-ice on.

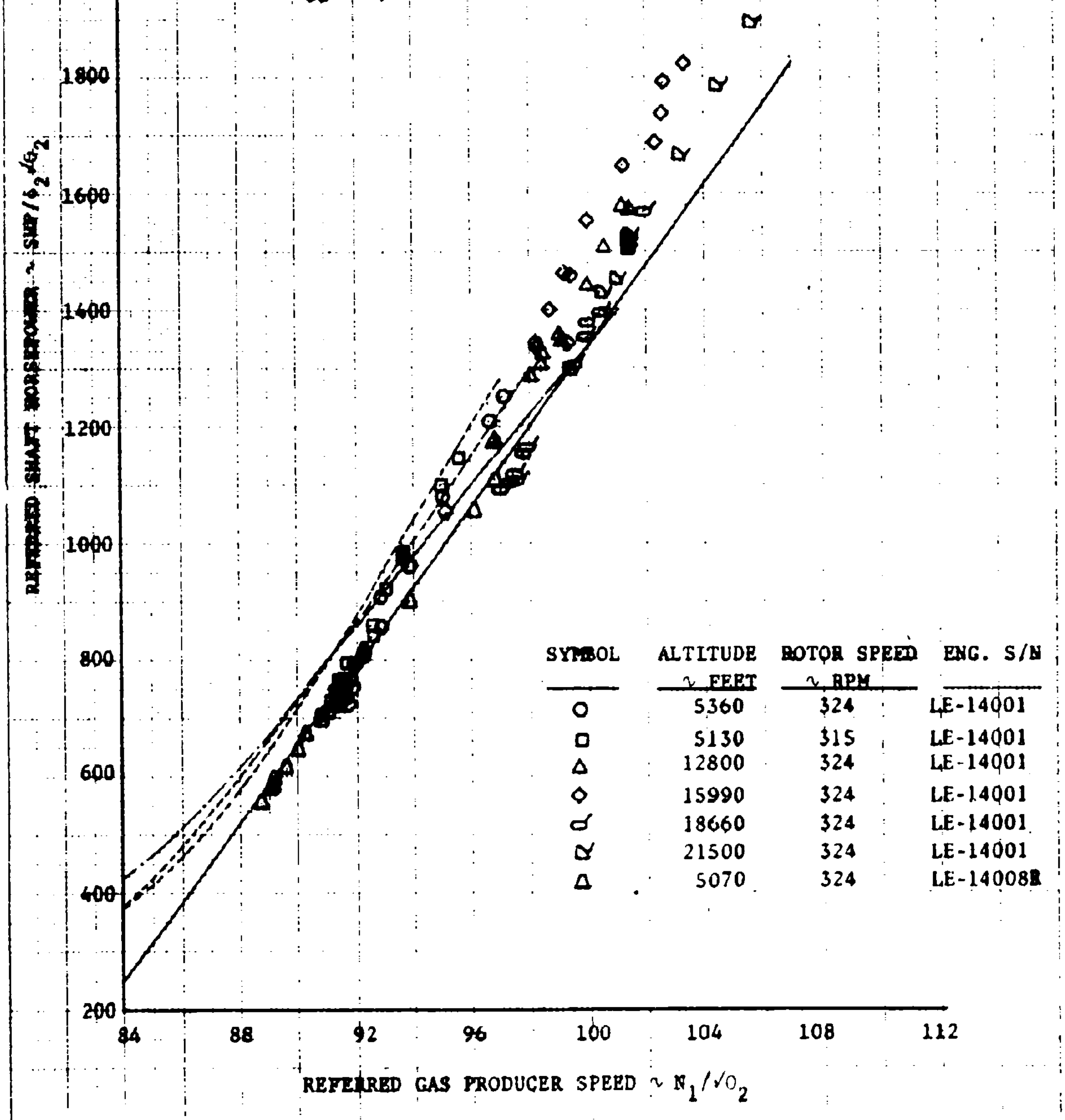


FIGURE 126
REFERRED EXHAUST GAS TEMPERATURE AND POWER
UH-1H T53-L-53

NOTE: Curve obtained from Model LTCK-4
Specification No. 104.33

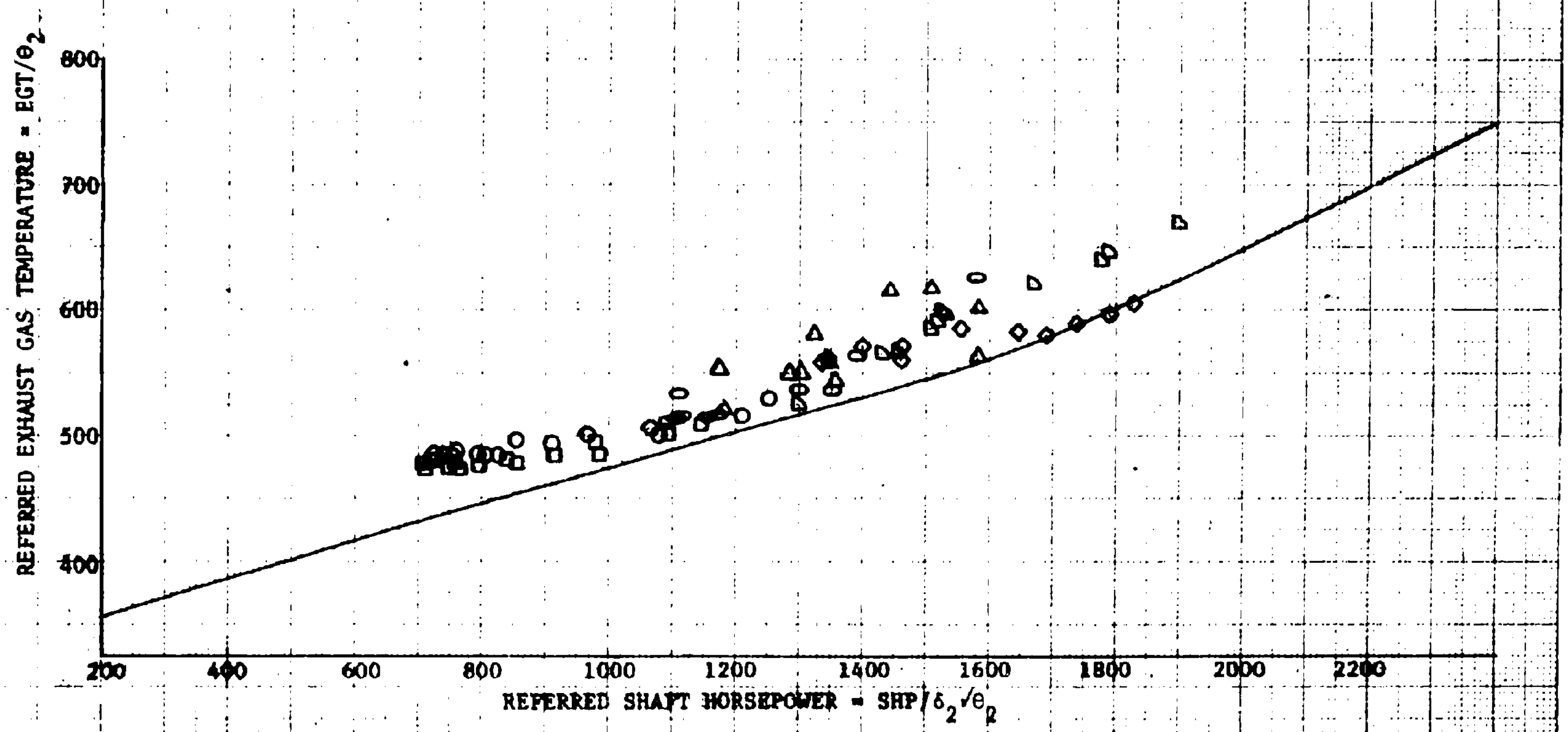


FIGURE 127

DEVIATION IN REGULATED GAS PRODUCER SPEED
(N_1 TOPPING SPEED)

T53-L-13

NOTE: 1. 100% N_1 = 25,150 RPM
2. Curve obtained from Engine manufacturer

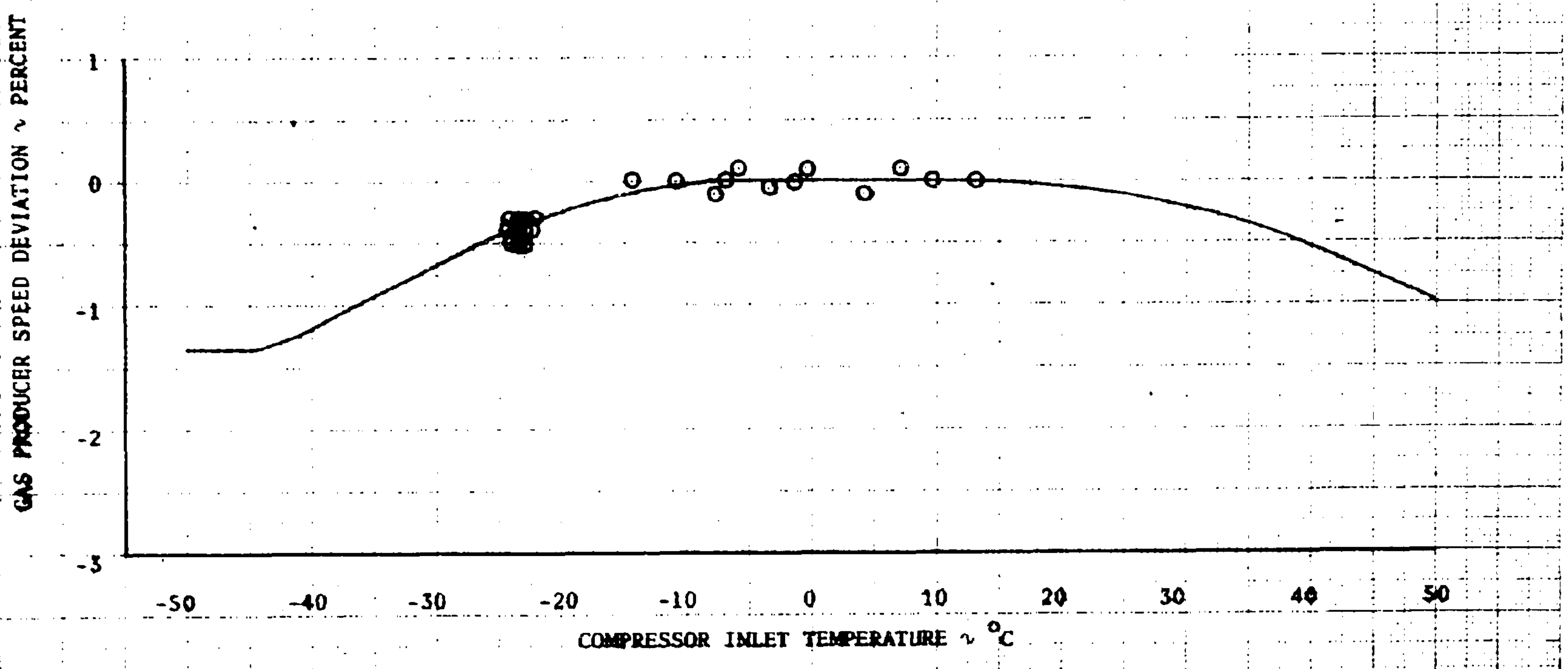


FIGURE 128
INLET GUIDE VANE SCHEDULE

T53-L-13

NOTES:

1. 100% N_1 = 25,150 RPM
2. Curves obtained from engine manufacturer
3. Engines may vary ± 1% from curves
4. Guide vanes move through 45° arc from full closed to full open

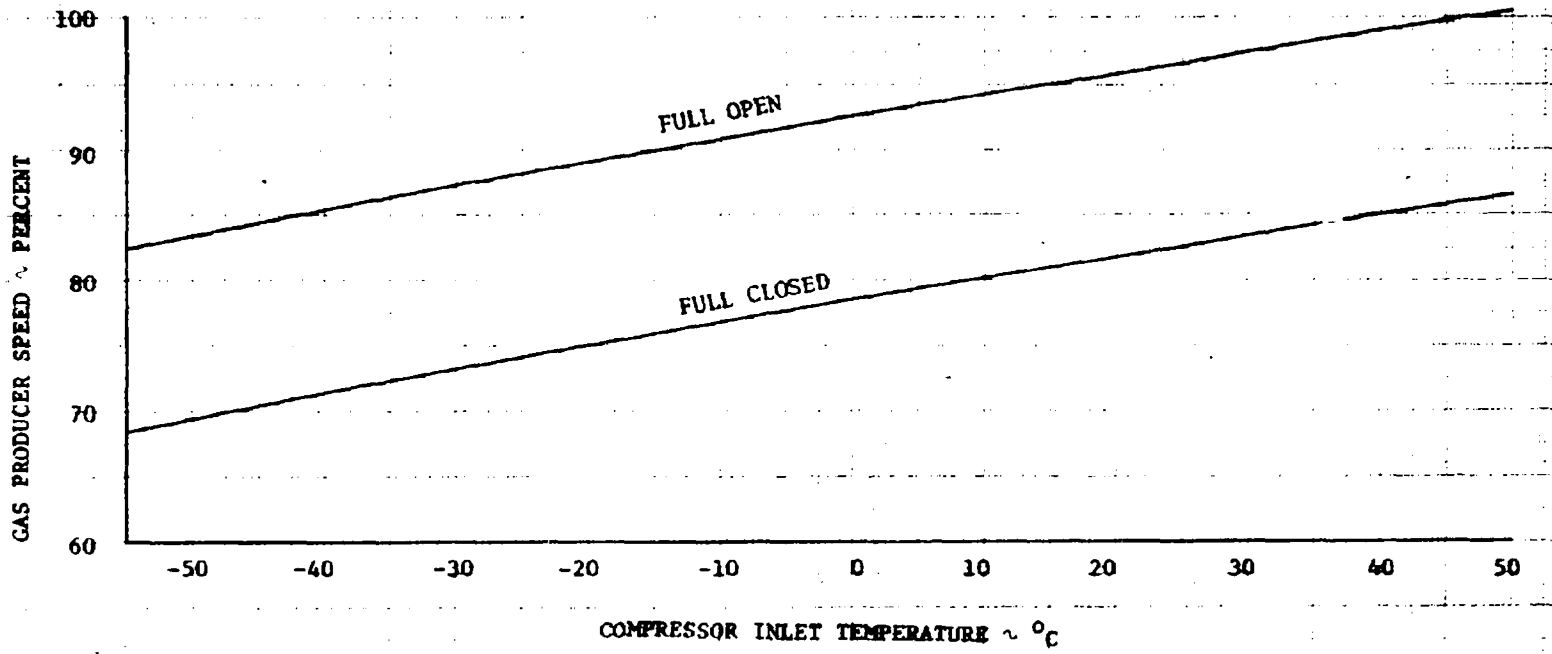


FIGURE 129
BLEED BAND SCHEDULE
T53-L-13

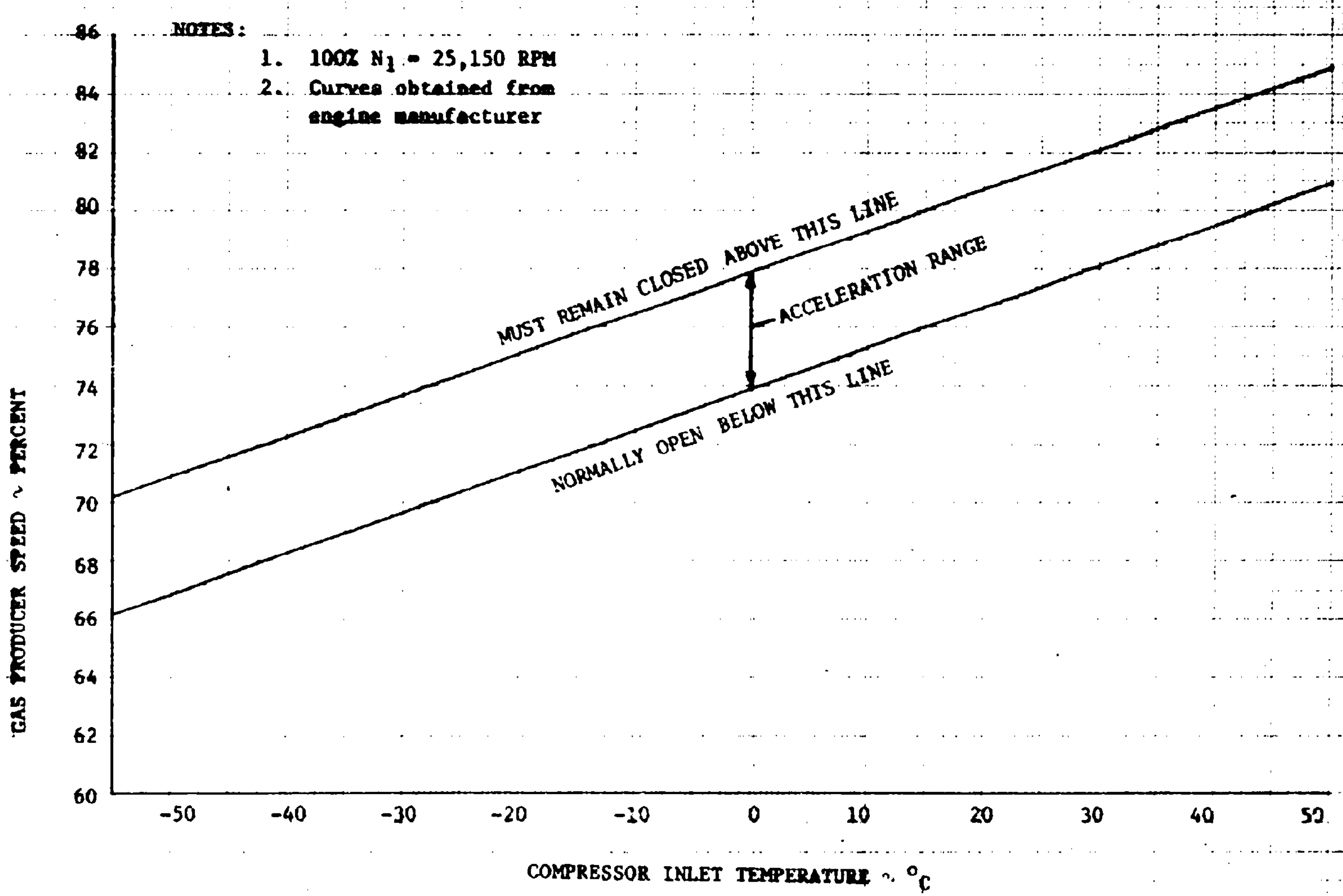
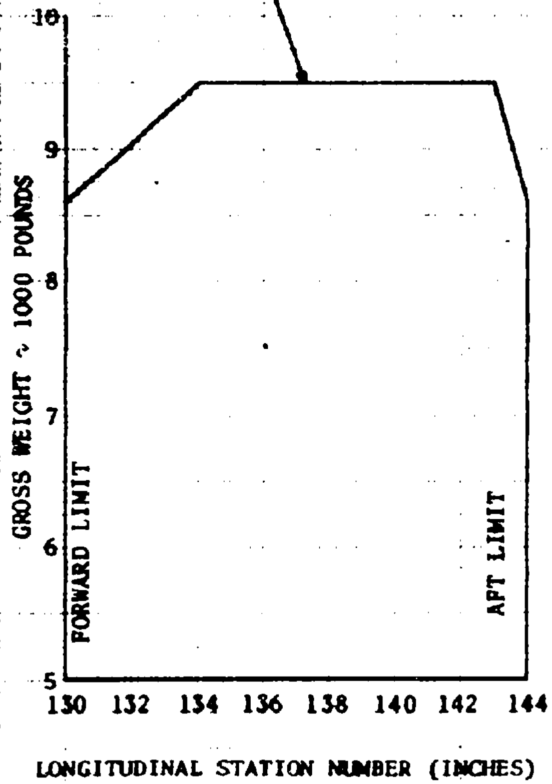


FIGURE 130
LOADING LIMITS
UH-1D AND UH-1H

PRESENT LIMITS

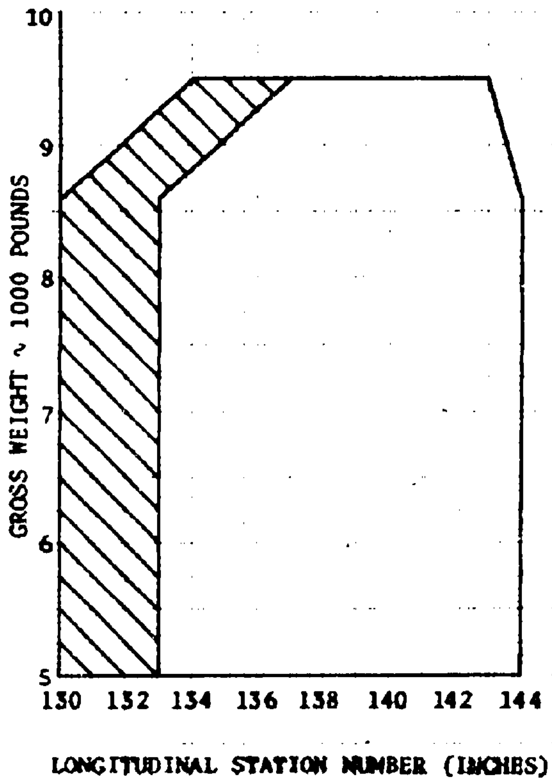


RECOMMENDED LIMITS

PRECAUTIONARY LOADINGS

INSUFFICIENT AFT STICK   INSUFFICIENT LEFT PEDAL

LANDINGS AND TAKEOFFS LIMITED TO ZERO WIND OR HEADWIND CONDITIONS FROM UNCONFINED AREAS. (FORMATION OR CONFINED AREA LANDINGS AND TAKEOFFS PROHIBITED)



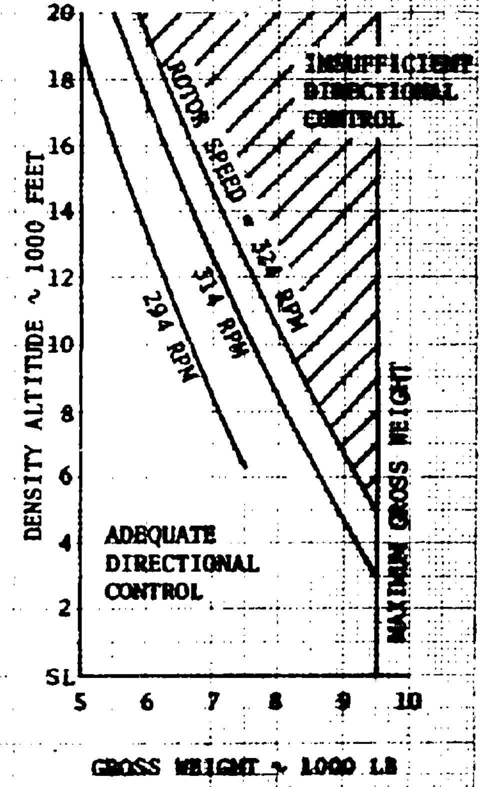
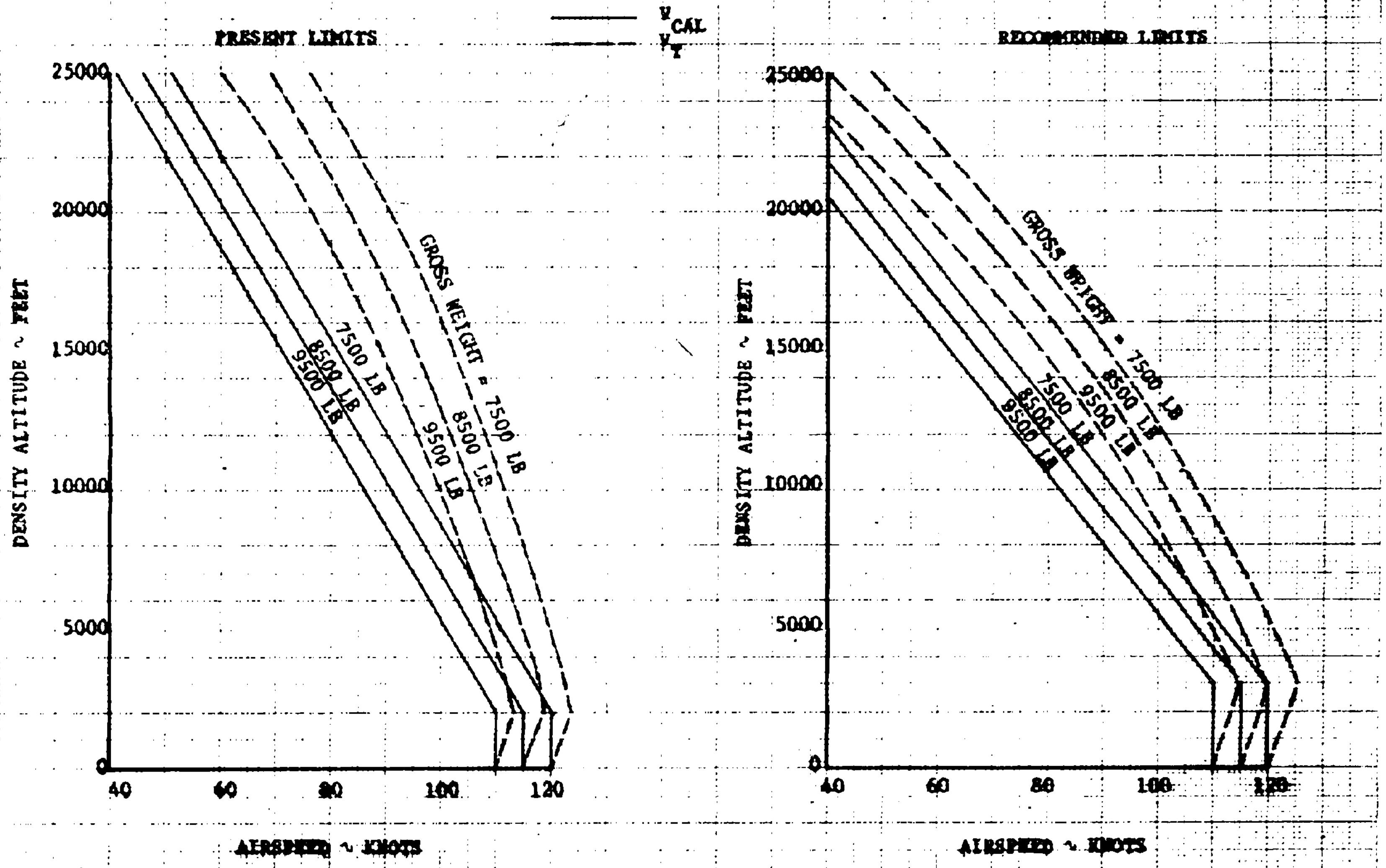


FIGURE 131
AIRSPEED LIMITS
UH-1D AND UH-1H



APPENDIX III. TEST TECHNIQUES AND DATA ANALYSIS METHODS

POWER DETERMINATION

1. Engine output power was determined from data obtained from sensitive torque pressure gages and rotor speed tachometers. Power was computed using the following equation:

$$SHP = \frac{2\pi \times T \times N_E}{33,000}$$

where: N_E = Output shaft speed (rpm)

T = Output shaft torque (ft-lb)

2. Calibration of the engine torque system indicated that torque was the following function of torque pressure:

$$T = C \times \Delta P \text{ (ft-lb)}$$

where: C = Constant (different for each engine), (ft-lb/in. Hg)

ΔP = Differential torquemeter pressure (in. Hg)

3. Differential torquemeter pressure was determined by subtracting the engine gear reduction housing internal pressure (P_{Lo}) from the hydromechanical torquemeter output pressure (P_{Hi}). Since a hydraulic system was used to measure torque, it was necessary to subtract the static hydraulic head from both the high and low torquemeter readings.

Considering these corrections, use the following equation:

$$\Delta P = (P_{Hi\text{-}ind} - P_{Hi\text{-}static}) - (P_{Lo\text{-}ind} - P_{Lo\text{-}static})$$

where: $P_{Hi\text{-}ind}$ = Indicated high torquemeter reading corrected for instrument error

$P_{Hi\text{-}static}$ = Static high torquemeter reading corrected for instrument error

$P_{Lo\text{-}ind}$ = Indicated low torquemeter reading corrected for instrument error

$P_{Lo\text{-}static}$ = Static low torquemeter reading corrected for instrument error

4. Engine speed may be determined from rotor speed as follows:

$$N_E = N_R \times 20.370 \text{ (rpm)}$$

By combining these equations, a convenient equation for determining output shaft horsepower is developed, as shown below:

$$SHP = \frac{2\pi}{33,000} \times 20.370 \times C \times N_R$$

$$(P_{Hi\text{-}ind} - P_{Hi\text{-}static} - P_{Lo\text{-}ind} + P_{Lo\text{-}static})$$

or

$$SHP = 0.003870 \times C \times N_R$$

$$(P_{Hi\text{-}ind} - P_{Hi\text{-}static} - P_{Lo\text{-}ind} + P_{Lo\text{-}static})$$

HOVER

5. Hover performance was obtained by the tethered and free-flight techniques. Tethered hover consists of restraining the helicopter to the ground by a cable and a load cell. Skid height was determined by varying the cable length. An increase in cable tension, measured by the load cell, had the same effect on hover performance as increasing gross weight. Free-flight hover tests consisted of stabilizing the helicopter at desired skid heights. Skid height was stabilized by the pilot while directed by a ground observer watching a weighted, measured cord suspended from the helicopter. All hover tests were conducted in winds of less than 2 knots.

6. Power required to hover was determined as explained in paragraph 1. For free-flight hover, thrust equaled gross weight. For tethered hover, thrust was determined as follows:

Thrust = Gross weight + load cell and cable weight + cable tension

7. All hovering data were reduced to nondimensional terms of C_P and C_T as follows:

$$C_P = \frac{SHP \times 550}{\rho A\ (\Omega R)^3}$$

where: SHP = Engine output shaft horsepower

ρ = Ambient air density (slugs/ft^3)

A = Rotor disc area (ft^2)

Ω = Rotor rotational speed (rad/sec)

R = Rotor radius (ft)

$$C_T = \frac{\text{Thrust}}{\rho A\ (\Omega R)^2}$$

All hover data were then plotted as C_P versus C_T for each skid height tested.

TAKEOFF

8. Takeoff performance was determined by the level acceleration technique. This technique involves a level acceleration, near the ground, from hover to the desired climbout airspeed. The rotation for climb was initiated 4 to 8 knots prior to reaching the desired climbout airspeed. Rotation of climb prior to reaching the climbout airspeed was necessary because of the aircraft attitude change required from that of acceleration to that of climbout. The reference skid height for hover and acceleration was 2 feet. A rotor speed of 324 rpm and maximum power available were used for each takeoff.

9. Takeoff tests were conducted to obtain data curves of climbout airspeed versus distance required to clear a 50-foot obstacle. Each curve was obtained by conducting a series of takeoffs using various climb airspeeds. During each series, ballast was added or removed as necessary to maintain the desired excess power available conditions as fuel was consumed and ambient temperature varied. A ground-operated Fairchild flight analyzer was used to produce a photographic record of time, horizontal distance and vertical distance for each takeoff. The climbout airspeed range used for each series of takeoffs varied from the minimum achievable to the maximum practical airspeed (approximately 50 KIAS). All takeoff tests were performed in winds of 2 knots or less.

10. The excess power method of takeoff analysis was used. C_P available was computed using the power available at the atmospheric conditions and rotor speed for the test. C_P required to hover at a skid height of 2 feet was determined by calculating

the C_T at takeoff conditions (weight, density altitude and rotor speed) and entering the 2-foot hover curve at this C_T to obtain the corresponding C_P. Then excess power was determined as follows:

$$\Delta C_p = C_p \text{ Available} - C_p \text{ Required at 2 ft}$$

11. Takeoffs were conducted at two field elevations (4200 and 11,700 ft) to verify ΔC_p method of takeoff analysis was valid for more than one altitude for this helicopter.

12. Distance to clear a 50-foot obstacle was obtained by plotting a time history of height and distance from the Fairchild plate and by reading the horizontal distance at the 50-foot skid height point. The distance was corrected for wind by multiplying the time required from takeoff initiation to 50 feet by the headwind component velocity and adding this result to the distance obtained from the Fairchild plate. The climb airspeed was determined from the height distance time history by calculating the horizontal and vertical velocities and then determining the resultant velocity along the flight path. The corrected distance was plotted versus climb true airspeed for each takeoff.

13. Takeoff performance was then summarized by combining all individual takeoff plots three dimensionally as distance required to clear a 50-foot obstacle versus ΔC_p and climb true airspeed. All dimensional takeoff performances were derived from this summary plot.

DECELERATION PERFORMANCE

14. Low-speed decelerations were made to determine both the distance required for aborting a takeoff and for power-on landing. Decelerations were made from stabilized speeds at heights of 2 and 50 feet. No rigid constraints were made on pilot technique. The pilot was instructed to make a "normal deceleration" as if aborting a takeoff. Rotor speed was set at minimum allowable steady state speed (314 rpm) prior to the deceleration, and no "beep" adjustment was made during the deceleration. The flare was executed to maintain a maximum steady state rotor speed of 324 rpm. Flight path time histories were recorded using a Fairchild flight analyzer camera. Data were reduced and are presented as distance to stop versus entry speed for different gross weights and entry heights.

CONTINUOUS CLIMBS

15. All continuous climb tests were flown at a constant rotor speed of 324 rpm with zero sideslip and using a constant power setting or UH-1H torque limit power, whichever was lower. When significant winds were present, climbs were flown on a crosswind heading to minimize wind effect. Climbs were flown either at optimum airspeed, as determined from level-flight data or sawtooth climb data, or at a constant 80 KCAS.

16. Continuous climb data were reduced and corrected to standard-day conditions as follows:

a. Test R/C was determined from pressure altitude variation with time and corrected for altimeter error caused by nonstandard temperature using the following equation:

$$R/C_{test} = \frac{dH_p}{dt} \times \frac{T_{A\ test}}{T_{A\ std}}$$

where: $\frac{dH_p}{dt}$ = Slope of pressure altitude versus time curve (ft/min)

$T_{A\ test}$ = Test-day absolute temperature at pressure altitude (°K)

$T_{A\ std}$ = Standard-day absolute temperature at pressure altitude (°K)

b. Test-day R/C was corrected for variations in power from test-day power available to standard-day power available at each density altitude using the following equation:

$$R/C_{power} = R/C_{test} + K_p \times 33{,}000 \times \frac{(SHP_{std} - SHP_{test})}{GRWT_{test}}$$

where: SHP_{std} = Standard-day shaft horsepower available determined from engine model specification 104.33 (ref 9, app I) corrected for installation losses (shp)

SHP_{test} = Shaft horsepower available during the climb (shp)

$GRWT_{test}$ = Test-day gross weight (lb)

K_p = Power correction factor determined from sawtooth climbs

c. Standard-day weight was determined by the following equation:

$$GRWT_{std} = GRWT_{SL\ std} - W_{f\ std} \times \frac{\Delta H_D}{60 \times Avg\ R/C_{power}}$$

where: $GRWT_{SL\ std}$ = Climb start gross weight at sea level on a standard day (lb)

$W_{f\ std}$ = Standard-day fuel flow determined from the engine model specification 104.33 (ref 9, app I) corrected for installation losses (lb/hr)

ΔH_D = Altitude increment (ft)

$Avg\ R/C_{power}$ = Average power-corrected rate of climb for altitude increment (ft/min)

d. The power-corrected R/C was corrected for test-day deviations in weight to obtain standard-day rate of climb using the following equation:

$$R/C_{std} - R/C_{power} + K_w \times 33{,}000 \times \frac{SHP_{std}\ (GRWT_{test} - GRWT_{std})}{GRWT_{test} \times GRWT_{std}}$$

where: K_w = Weight correction factor determined from sawtooth climbs

17. After the standard R/C was obtained, time to climb, distance traveled and fuel used were obtained using the following equations:

$$\text{Time to climb} = \frac{\Delta H_D}{Avg\ R/C_{std}}\ (\text{min})$$

$$\text{Distance} = \frac{\Delta H_D}{60 \times Avg\ R/C_{std}} \times (V_T)^2 - (.9875 \times R/C_{std})^2\ (\text{NM})$$

$$\text{Fuel used} = \frac{\Delta H_D}{60 \times Avg\ R/C_{std}} \times W_{f\ std}\ (\text{lb})$$

where: $Avg\ R/C_{std}$ = Average standard R/C for altitude increment (ft/min)

18. R/C standard, time to climb, shaft horsepower standard, distance traveled, weight, fuel used, calibrated airspeed and true airspeed were plotted as functions of standard altitude and are presented in appendix II.

CLIMB SUMMARIES

19. Climb summaries were derived from level-flight data and verified by continuous climb data. Power required for level flight was obtained from the nondimensional level-flight data at the desired specific conditions of weight, altitude, temperature, rotor speed and airspeed. R/C was then determined at those conditions as follows:

$$R/C = \frac{K_p \times 33{,}000 \times (SHP_{avail} - SHP_{LF})}{GRWT}$$

where: K_p = Power correction factor determined from sawtooth climbs

SHP_{avail} = Engine output shaft horsepower available at specific conditions of interest (shp)

SHP_{LF} = Engine output shaft horsepower required for level flight at the specific conditions of interest (shp)

GRWT = Helicopter gross weight (lb)

20. Time to climb, distance traveled and fuel used were determined after determining R/C as explained in paragraph 17.

CLIMB CEILING SUMMARIES

21. Climb ceiling summaries were derived from level-flight data and verified by continuous climb data. The increment of power required for the appropriate R/C (100 or 500 fpm) was determined as follows:

$$\Delta SHP = \frac{R/C \times GRWT}{K_p \times 33{,}000}$$

22. This incremental power was added to the level-flight power required at specific flight conditions. The particular ceiling was then defined as the altitude where this total power was equal to the power available.

SAWTOOTH CLIMBS

23. Sawtooth climbs were flown to determine the effect on climb performance, of gross weight, rotor speed and power. Sawtooth climbs were also flown to determine optimum climb speed, up to service ceiling. All parameters except the one of interest were either maintained constant or corrected to a constant value. Generally, all sawtooth climbs of a set were flown by maintaining a stabilized climb through the same 1000-foot altitude band. The altitude band was decreased near service ceiling where R/C was low. Test rate of climb was computed as shown in paragraph 16.

Airspeed Variation

24. Optimum climb (maximum R/C) airspeed was determined from level-flight data at minimum power required for low altitudes and light weight. At high altitudes and/or heavy weights, optimum climb speed was determined by flying several sawtooth climbs at airspeeds approximately 5 knots apart. It was necessary to correct all gross weights of a set to the average gross weight to eliminate the variation in R/C caused by weight effect. The weight correction equation given in paragraph 16 was used. All other parameters were held constant. It was found that optimum climb true airspeed was an explicit function of C_T but not necessarily the speed for minimum power.

Rotor Speed Variation

25. Rotor speeds of 314, 319 and 324 rpm were used during the investigation of the effect of weight variation on climb performance. A rotor speed of 324 rpm was determined to produce the maximum R/C at most conditions.

Weight Effect

26. Sawtooth climbs were flown at various gross weights for different sets of altitude, weight, power and rotor speed conditions at the optimum climb airspeed (para 24). A weight correction factor was then determined as follows:

$$K_W = \frac{\Delta R/C}{\Delta GRWT} \times \frac{GRWT^2}{SHP \times 33{,}000}$$

where: $\frac{\Delta R/C}{\Delta GRWT}$ = Slope of rate of climb versus gross weight curve

GRWT = Gross weight at which slope was read (lb)

SHP = Power at which sawtooths were flown

The weight correction factor (K_W) was found to be an explicit function of C_T for a given rotor speed.

Power Effect

27. Sawtooth climbs were flown at various powers for different sets of altitude, weight and rotor speed conditions at the optimum climb airspeed. It was necessary to correct all gross weights of a set of conditions to the average grwt to eliminate the variation in R/C due to weight effect. A power correction factor was then determined as follows:

$$K_p = \frac{\Delta R/C}{\Delta SHP} \times \frac{GRWT}{33,000}$$

where: $\frac{\Delta R/C}{\Delta SHP}$ = Slope of rate of climb versus shaft horsepower curve

GRWT = Corrected gross weight (lb)

The power correction factor (K_p) was found to be approximately a constant 0.80.

LEVEL FLIGHT

28. Level-flight performance tests were flown to determine power required for level flight, engine fuel flow, power characteristics, airspeed limits, range, and endurance. Each flight was flown at a constant C_T. This technique required that GRWT/ρ be held constant by increasing altitude as gross weight decreased. Engine parameters, airspeed, fuel used, and atmospheric conditions were recorded for each test point. Gross weight, density altitude and rotor speed combinations were used to cover the desired C_T range. Power required was based on the installed engine torquemeter and rotor speed utilizing the Lycoming engine calibration. Horsepower was corrected to standard altitude by the following equation:

$$SHP_s = SHP_t \times \frac{\rho_s}{\rho_t}$$

where: SHP_s = Corrected shaft horsepower (shp)

SHP_t = Test shaft horsepower (shp)

ρ_s = Standard ambient air density based on average density altitude for the flight (slugs/ft^3)

ρ_t = Ambient air density for each test point (slugs/ft^3)

29. The standard gross weight was calculated for the standard density and average C_T using the equation:

$$W_s = C_{T\ average} \times \rho_s \times A\ (\Omega R)^2$$

where: $C_{T\ average}$ = Average C_T for the flight

A = Rotor disc area (ft^2)

Ω = Rotor angular velocity (rad/sec)

R = Rotor radius (ft)

30. The nondimensional parameters C_T, C_P, μ and Mach number were used to correlate the level-flight data. Airspeeds and power required for each flight were reduced to the nondimensional parameters, and C_P versus μ was plotted for the average C_T. The tip Mach number of the advancing blade was calculated for each point using the equation:

$$M = \frac{V_T + 0.592\ (\Omega R)}{38.967\ T_{a_t}} = \frac{(1 + \mu)(0.592\ \Omega R)}{38.967\ T_{a_t}}$$

where: M = Advancing tip Mach number

V_T = Aircraft true airspeed (kt)

T_{a_t} = Test absolute ambient temperature (°K)

μ = Advance ratio (ratio of aircraft speed to rotor tip speed)

31. The data were analyzed and summarized completely in four-dimensional form with only the condition that it vary continuously in each of the three planes of μ-C_P, M-C_P and C_T-C_P and that the summary match the test data. This summary was then used to derive the range and endurance summaries.

DYNAMIC STABILITY

32. The dynamic stability of the UH-1H was qualitatively determined by observing the helicopter behavior following an artificial disturbance. This artificial disturbance was the result of a pulse-type control input about the axis being investigated. The pulse input

was made by rapidly displacing the control approximately 1 inch from trim position, holding for approximately 1 second and then rapidly returning to trim position.

STATIC STABILITY

33. The static lateral and directional stability of the UH-1H was qualitatively determined by observing the lateral cyclic and directional control positions during steady state sideslip. The angle of sideslip was investigated to approximately 15 degrees in both directions. The lateral cyclic control position defined the static lateral stability or dihedral effect. The directional control position defined the static directional stability.

APPENDIX IV. DETAILED HELICOPTER DESCRIPTION

SOURCES OF INFORMATION

1. The information contained in this appendix was obtained from the operator's manual (ref 12, app I), the airframe model specification (ref 17), the engine model specification (ref 9), the airframe manufacturer, the engine manufacturer or directly from measurement of the test aircraft.

DESIGN DATA

2. The design data is listed as follows:

Overall Dimensions

Length (rotor turning)	57 ft, 1.1 in.
Length (nose to tail)	41 ft, 11.1 in.
Width of skids (maximum width except rotor)	9 ft, 6.6 in.
Height (to top of turning tail rotor)	14 ft, 5.5 in.
Height (to top of rotor mast)	14 ft, 0.7 in.
Fuselage ground clearance (at design weight)	1 ft, 3.0 in.
Main rotor clearance (rotor tip to tail boom, static)	1 ft, 10.7 in.

Weights

Manufacturer's empty weights	4973 lb
User's empty weight (see para 125)	5400 lb
Design gross weight	6600 lb
Maximum overload gross weight	9500 lb

Main Rotor

Number of blades	2
Rotor diameter (blades)	48 ft

Rotor diameter (including tracking tips)	48 ft, 3.2 in.
Blade chord (root to tip)	21 in.
Blade airfoil (root to tip)	NACA 0012
Blade twist (root to tip)	-10 deg
Preconing angle	2.75 deg
Mast angle (relative to horizontal reference)	5 deg forward tilt
Control travel: (measured at center of grip)	
Collective	10.75 in. (27 deg)
Longitudinal cyclic	12.2 in (30 deg)
Lateral cyclic	12.3 in. (30 deg)
Blade travel:	
Flapping (any direction)	±11 deg
Collective (measured at 75% radius)	0 to 15 deg
Longitudinal cyclic	±12 deg
Lateral cyclic	±10 deg

Tail Rotor

Number of blades	2
Rotor diameter	8 ft, 6 in.
Blade chord (root to tip)	8.41 in.
Blade airfoil (root to tip)	NACA 0015
Pedal travel	6.8 in.
Blade travel:	
Thrust to right (left yaw)	+19 deg
Thrust to left (right yaw)	-7 deg

DERIVED DATA

Main Rotor

Disc area (total swept area)	1809 ft^2

Blade area (including hub)	82 ft^2
Solidity	0.0464
Disk Loading:	
6600 lb	3.65 lb/ft^2
9500 lb	5.25 lb/ft^2
Blade loading:	
6600 lb	80.5 lb/ft^2
9500 lb	115.9 lb/ft^2
Power loading: (1137 shp)	
6600 lb	5.80 lb/shp
9500 lb	8.36 lb/shp
Tip speed in a hover:	
324 rotor rpm (maximum)	814.3 fps (482.1 kt)
294 rotor rpm (minimum)	738.9 fps (437.5 kt)
Maximum tip speed in forward flight: (V_T - 123.6 kt)	
Power on (324 rotor rpm)	1023.0 fps (605.7 kt)
Power off (339 rotor rpm)	1068.0 fps (628.1 kt)

Tail Rotor

Disk area (total swept area)	56.7 ft^2
Blade area (including hub)	5.96 ft^2
Solidity	0.105
Tip speed in a hover:	
324 rotor rpm	736 fps (436 kt)
294 rotor rpm	668 fps (395 kt)

Gear Ratios

Power turbine to output shaft	3.2105:1
Output shaft to main rotor	20.370:1
Output shaft to tail rotor	3.990:1
Gas producer turbine to tach pad (100% = 25,150 rpm)	5.988:1

FLIGHT LIMITATIONS

3. The limits contained in this section were obtained from Army manuals. Various recommendations to change these limits are made in this report.

Engine and Drive Train

Power ratings:	
Military power (30-minute limit)	1400 shp derated to 1100 shp
Maximum continuous power	1250 shp derated to 1100 shp
Torque limits:	
Maximum continuous	50 psi
Transient overtorque (not to be used intentionally) (no maintenance required)	50 to 54 psi
Transient overtorque (inspect drive train)	54 to 61 psi
Transient overtorque (replace all drive train and rotor components)	Over 61 psi
Output shaft speed:	
Maximum steady state	6600 rpm
Minimum steady state	6400 rpm
Minimum steady state below 7500 lb	6000 rpm
Maximum transient (below 91% N_1) (not to be used intentionally)	6750 rpm

Exhaust Gas Temperature

Maximum continuous	390° to 625°C
30-minute limit	625° to 645°C
5-second limit for starting and acceleration	675°C
Maximum for starting and acceleration	760°C

Gas Producer

Maximum speed	25,600 rpm (101.8%)
Flight idle speed	15,900 to 17,000 rpm (63 to 68%)
Ground idle/start speed	12,100 to 13,100 rpm (48 to 52%)

Rotor Speed

Maximum power on	324 rpm
Power on transient	331 rpm
Power off	339 rpm
Minimum power on	314 rpm
Power on less than 7500 lb	294 rpm
Power off	294 rpm

Airframe

Loading:	
Design weight	6600 lb
Maximum overload weight	9500 lb
Maximum floor loading	300 lb/ft^2
Maximum cargo hook capacity	4000 lb
Maximum forward cg	Sta 130
Maximum aft cg	Sta 144
Maximum lateral cg (see fig. 130, app II, for complete cg envelope)	±7.5 in.
Limit load factors:	
Positive 6600 lb	+3.0 g's
9500 lb	+2.1 g's
Negative 6600 lb	-0.5 g
9500 lb	-0.35 g
Airspeed:	
Forward flight	
Maximum (see fig. 131, app II, for complete airspeed envelope)	123.6 KTAS at 2000 ft
Sideward and rearward flight Maximum	30 kt

ENGINE MODIFICATIONS

4. Major design changes made to the T53-series engines to arrive at the T53-L-13 configuration are as follows:

a. A second gas producer turbine stage was added.

b. A second power turbine stage was added.

c. Variable inlet guide vanes were added.

d. The axial compressor was modified.

e. The fuel control was modified.

f. The output gear ratio was changed from 3.2057:1 to 3.2105:1.

These modifications produced the following changes:

a. Increased weight (34 to 530 lb).

b. Improved acceleration.

c. Increased airflow.

d. Increased power from 1100 to 1400 shp.

e. Decreased specific fuel consumption at powers above 700 referred shp.

INLET CONFIGURATIONS

5. Five different combinations of air inlet devices (two engine mounted and three airframe mounted) were investigated to determine their effect on engine performance. These inlet devices are described below.

Bellmouth

6. The bellmouth is mounted directly to the compressor housing. It provides no air filtration. Its purpose is to provide smooth air flow to the compressor. When no external filters are used, the bellmouth is covered with a large mesh screen.

Particle Separator

7. When used, the particle separator replaces the bellmouth. In addition to providing smooth air flow to the compressor, it separates sand and dust particles from the inlet air. All engine inlet air passes through the particle separator. Approximately 5 percent of the air flow, along with any sand or dust particles, is inertially separated from the main air flow. This 5-percent air flow then

continues to the compressor. The design also provides for cooling air to flow around the short shaft coupling. Photo A shows the particle separator mounted on the engine. Photo B shows the particle separator partially disassembled.

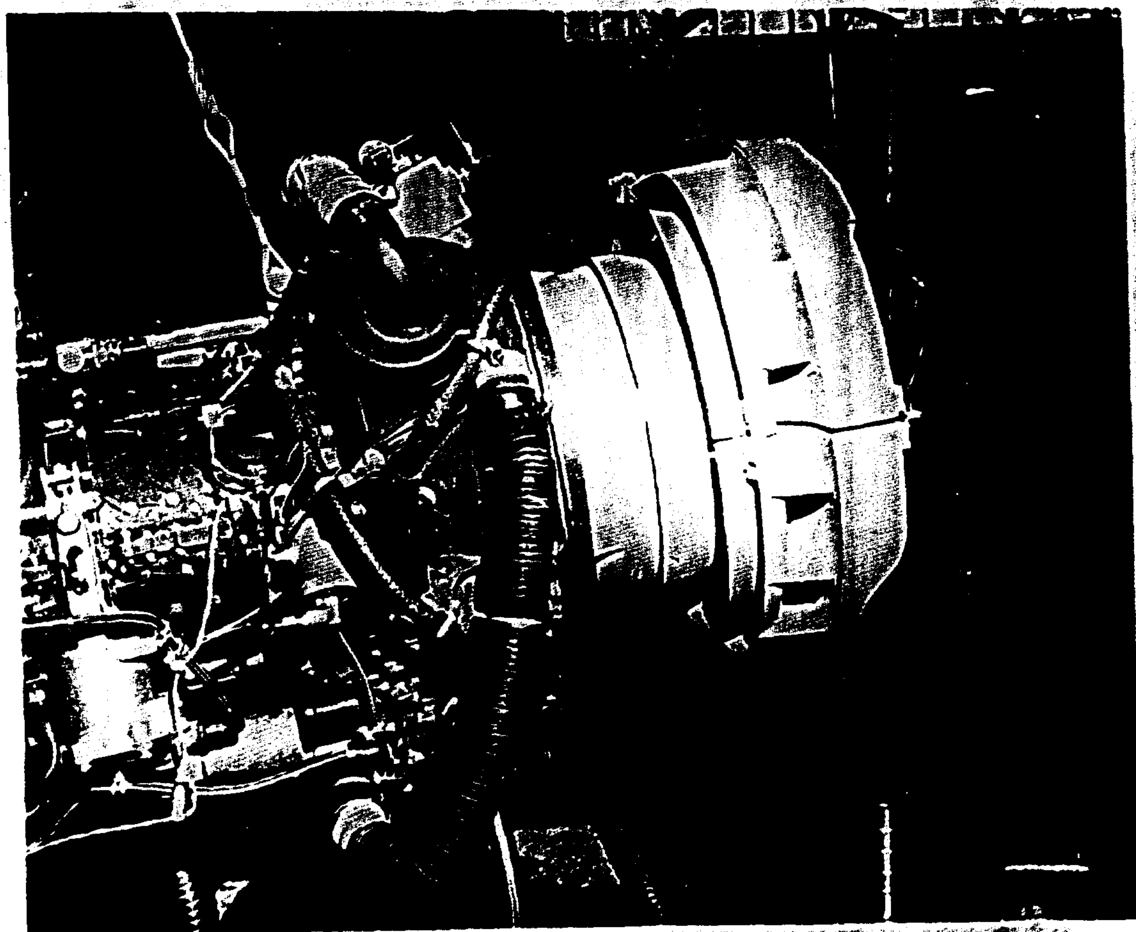


Photo A. Particle Separator Mounted on Engine.

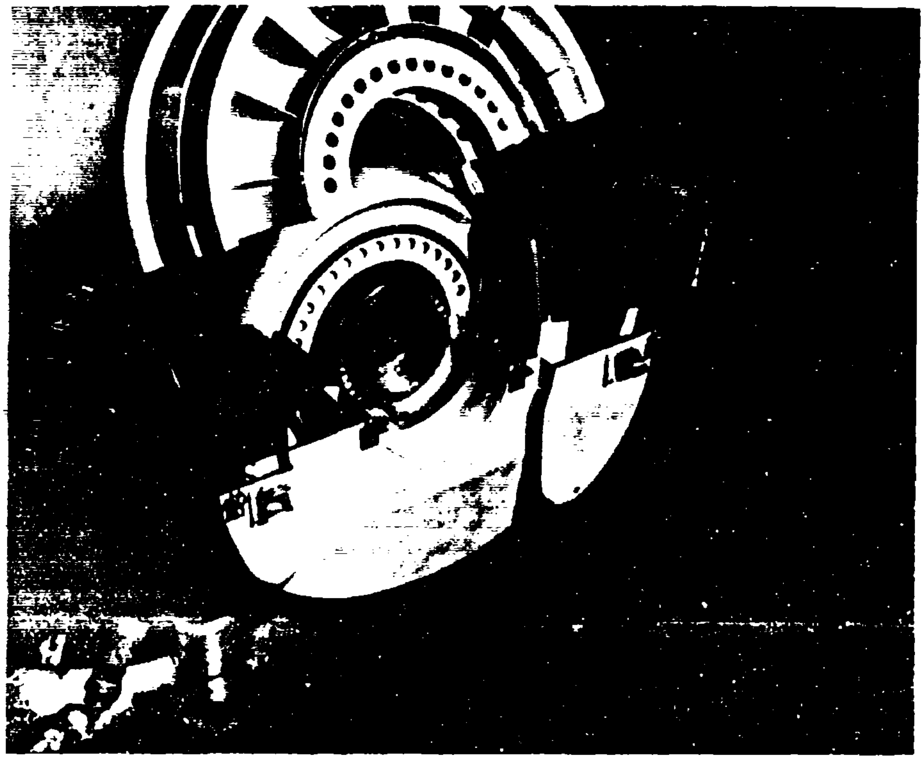


Photo B. Particle Separator Partially Disassembled.

Barrier Filter Installation

8. The barrier filter is mounted on the outside of the air inlet plenum chamber (photo C). It consists of a double-wall, corrugated, large mesh screen filled with a fibrous material. The filler material physically blocks sand and dust particles. Provisions are made to open up the top portion of the filter if it becomes blocked with ice, etc.

Photo C. Barrier Filter Installation.

External Screen

9. An external screen was used to simulate the barrier filter with the filler material removed. This is sometimes done when the barrier filter becomes excessively dirty or deteriorated.

External Louvers

10. The external louvers are used in place of the barrier filter on some UH-1H/D helicopters. They provide no air filtration but do permit partial ram air pressure recovery in forward flight. The louvers are shown in photo D.

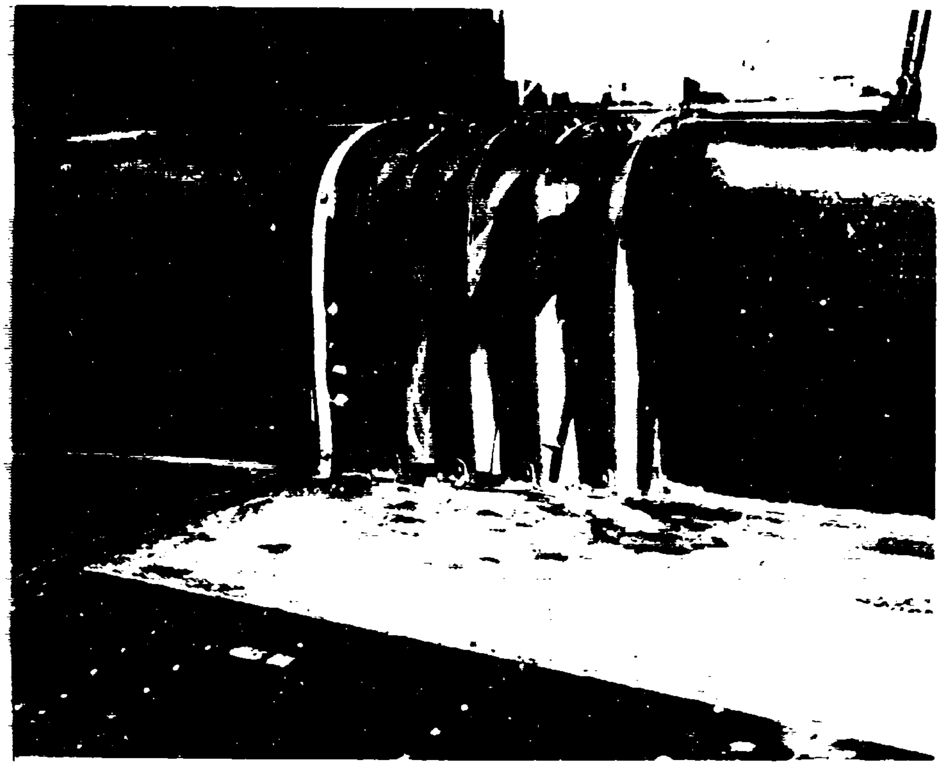


Photo D. External Louvers.

TEST AIRCRAFT CONTROL RIGGING

11. Prior to these tests, the aircraft control rigging was checked. The swash plate and synchronized elevator were not within allowable tolerances. The aircraft was rerigged as follows:

Item	Nominal Rigging (deg)	Tolerance (deg)	Test Rigging (deg)
Swash plate (relative to mast):			
Longitudinal full forward	-12.0	±0.5	-11.8
Longitudinal full aft	+12.0	±0.5	+12.0
Lateral full left	-10.0	±0.5	-10.5
Lateral full right	+10.0	±0.5	+9.5
Pedal (tail rotor):			
Full left	+19.0	±1.0	+19.7
Full right	-7.0	±1.0	-7.2
Synchronized elevator:			
Full forward stick	[1]+3.75	±0.5	[1]+3.75
Full aft stick	[1]+2.5	±1.0	[1]+2.5
Max nose down	[2]-1.5	±0.5	[2]-1.5
Mast angle forward tilt	5.0	None	5.1

[1]Nose up.
[2]Nose down.

TEST AIRCRAFT WEIGHT AND BALANCE

12. The test aircraft (S/N 60-6029) was weighed with the fuel tanks drained and all other fluid reservoirs full. With all test instrumentation installed, the weight and balance was as follows: weight, 5293 pounds; longitudinal cg station, 144.86; lateral cg station, 0.38-inch right. With all instrumentation removed, the weight and balance was as follows: weight, 4978 pounds; longitudinal cg station, 146.50; and lateral cg station, 0.0. Since this was a prototype aircraft and configured for test purposes, it was not considered to be representative of normal UH-1H aircraft.

ADDITIONAL INFORMATION

13. Additional descriptive information on aircraft systems and other details can be obtained from references 9, 12 and 17, appendix I.

APPENDIX V. TEST INSTRUMENTATION

GENERAL

1. The test instrumentation used for this evaluation was supplied, installed and maintained by the Logistics Division of the US Army Aviation Systems Test Activity. A swivel-mounted pitot static airspeed head, incorporating angle-of-attack and sideslip vanes, was installed on a nose boom and was mounted approximately 8 feet forward of the nose of the helicopter. The static pressure ports of this pitot static head were the pressure source for the sensitive altimeter, as well as the sensitive boom airspeed indicator.

PILOT/ENGINEER'S PANEL

2. The pilot/engineer's panel is shown in photo I. Sensitive calibrated instrumentation was installed on the pilot/engineer's panel to measure the following parameters:

Boom system airspeed
Boom system altitude
Free air temperature
Gas producer speed
Low torquemeter pressure
Total fuel used
Collective stick position
Cockpit normal acceleration
Bleed bank position
Standard system airpseed
Angle of sideslip
Rotor speed
High torquemeter pressure
Exhaust gas temperature
Fuel flow rate
Longitudinal stick position
Photopanel frame counter
Aircraft heading

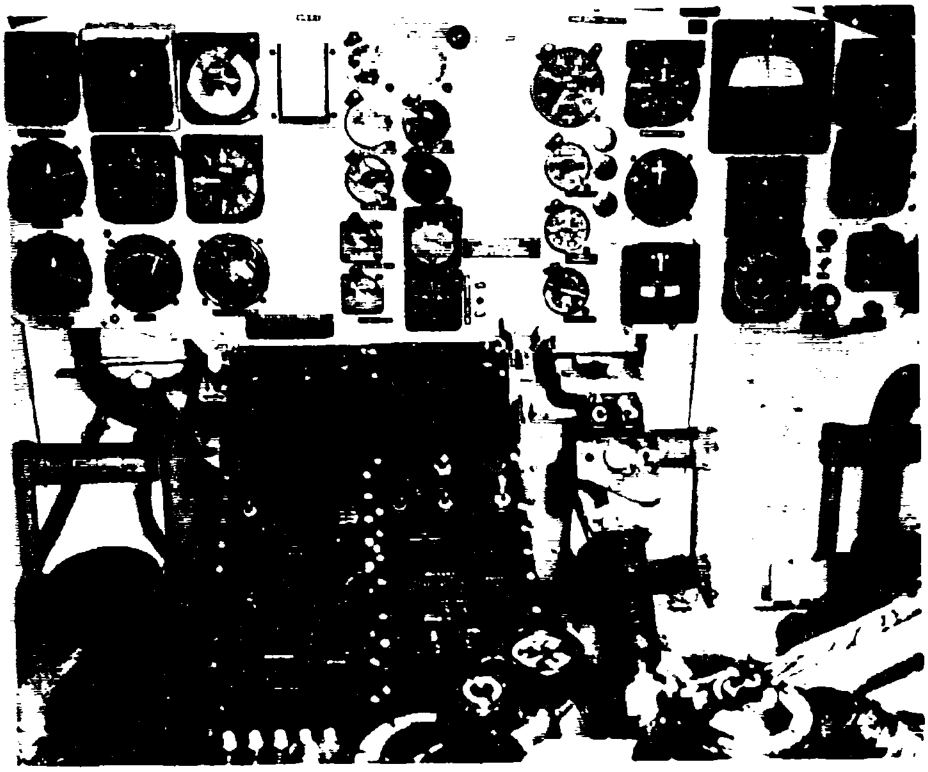


Photo I. Cockpit Instruments Pilot/Engineer's Panel.

PHOTOPANEL

3. The photopanel instrumentation is shown in photo II. Sensitive calibrated instrumentation was installed in the photopanel to measure the following parameters:

Boom airspeed	Free air temperature (dial)
Boom altitude	Free air temperature (digital)
Compressor inlet pressure	Compressor inlet temperature (dial)
Compressor discharge pressure	Compressor inlet temperature (digital)
Exhaust gas temperature	N_2 speed select "Beep" position indicator
Engine compartment pressure	
Longitudinal stick position	

Collective stick position	Lateral stick position
Angle of attack	Pedal position
Stopwatch time	Time of day
Photo frame number	Fuel used
Gas producer speed	Guide vane position
Rotor speed	High torque pressure
Throttle position	Low torque pressure
Fuel control output pressure	Bleed band position
Responsive torque pressure	

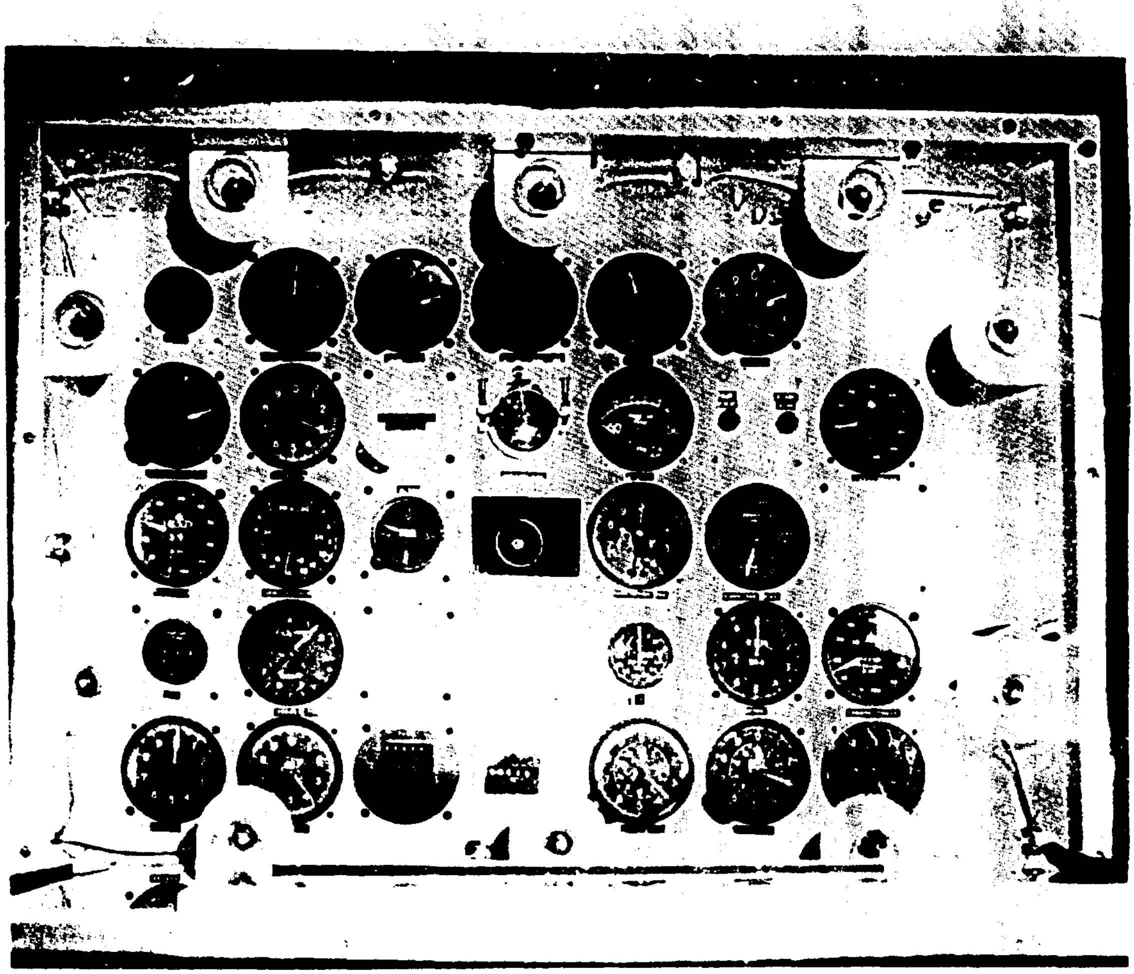


Photo II. Photopanel Instruments.

APPENDIX VI. MAINTENANCE

GENERAL

1. The YUH-1H test helicopter, S/N 60-6029, was a prototype aircraft which has been used exclusively for test purposes since it was produced. As such, it probably cannot be considered to be representative of present production UH-1H helicopters from a maintenance standpoint.

PARTS CONSUMPTION

2. Parts consumption of the test aircraft was considered to be relatively high. Approximately 4 weeks of testing were lost during the flying portion of this program while awaiting delivery and/or installation of parts which required replacement. Table A lists parts replaced during the program or shortly after its completion.

MISCELLANEOUS

3. Many components which were not replaced required repair, rework or adjustment. Continual problems were experienced with the EGT system. Although the feathering bearing grip seals were changed prior to the program because of leakage, they continued to leak during the program. The engine air inlet filters added significantly to aircraft maintenance. They required daily cleaning. Approximately ½ hour was required to partially disassemble the particle separator and clean the internal filter. Approximately 15 minutes were required to remove and clean the barrier filter. Future air inlet filters should be designed to minimize or eliminate cleaning requirements.

Table A. Parts Replacement

Component	Discrepancy	Airframe Time
Airscoop assy	Bent	537:30
Airspeed indicator	Out of adjustment	537:30
Rivet, tail boom	Loose	537:30
Swashplate bearing	Worn excessively	537:30
EGT gage	Fluctuates	537:30
Thermocouple harness	Erratic	538:10
Damper assy	Out of adjustment	540:45
Mixing tube	Scored	540:45
Rod assy	Bent	540:45
Catch, flush cowel	Broken	559:45
Seal, main cargo LH door	Torn	562:30
Seal, main cargo RH door	Torn	564:25
Roller cam	Worn excessively	565:25
Support hanger, roller	Worn excessively	566:10
Light, panel assy caution	Burned out	576:20
Servo cylinders	MWO compliance	649:00
Trunion assy, swashplate	Worn excessively	649:00
Bearing, ball airframe	Worn excessively	649:00
Slider, T/R	Wrong part	666:00
Slider nuts, T/R	Loose	669:00
Support door handle (2 ea)	Worn excessively	672:00
Elevator assy, RH	Fluctuates, unstable	672:00
Elevator assy, LH	Fluctuates, unstable	672:00
Link assy, T/R	Worn excessively	672:00
Support assy elevator	Worn excessively	672:00
Connect link	Loose rivets	672:00
Harness A/C shoulder	Removed for time change	672:00
Elbow assy	Buckled	672:00
Latch assy (2 ea)	Worn excessively	672:00
Compressor blade	Chipped	672:00
Ignition plugs	Flaking	672:00
Separator seal	Torn	673:45
Hydraulic reservoir	Leaking	673:45
Seat covers	Worn excessively	699:05
Fuel manifold flow divider	Leaking	705:55
Fuel oil screens	Dirty	762:20
Aft closs tube rubber bumper	Missing	762:20
Drive shaft coupling, T/R	Leaking	762:20
Hydraulic reservoir	Leaking	762:20
Light bulb (4 ea)	Burned out	765:50
Hydraulic system	Leaking	767:35
ADF dynamotor	Burned out	775:50
Windshield wiper	Worn excessively	778:25
Cable, T/R	Worn excessively	778:25
Roller, cabin door	Worn excessively	778:25
Fastener	Missing safety wire	779:55
Seal, N_1 tach generator	Leaking	798:00
Bushings, gimbal ring	Loose	808:00
Scissors and sleeve	Loose	808:00
Transmission friction damper	Loose	812:00
Spring, pilot door hinge	Broken	812:00

APPENDIX VII. DEFINITIONS OF SYMBOLS AND ABBREVIATIONS

Symbol	Definition
A	Area; square feet or square inches
a	Speed of sound; usually in feet per second
C_P	Power coefficient; a nondimensional unit of power $$C_P = \frac{550\ SHP}{\rho A\ (\Omega R)^3}$$
C_T	Thrust coefficient; a nondimensional unit of thrust or weight $$C_T = \frac{W}{\rho A\ (\Omega R)^2}$$
H_D	Density altitude; a measure of air density expressed in feet above sea level
H_p	Pressure altitude; a measure of air pressure expressed in feet above sea level
ic	Instrument corrected, used as a subscript
°K	Degrees Kelvin; a unit of temperature
k_p	Power efficiency factor; usually used for climb corrections
K_w	Weight correction factor; usually used for climb corrections
M	Mach number, ratio of velocity to speed of sound
M_{AT}	Main rotor advancing tip mach number
N, n	Rotational speed; usually in revolutions per minute
N_R	Rotational speed of rotor
P_{AMB}, P_{amb}, P_0	Ambient pressure

Symbol	Description
P_{T_2}, P_2	Total compressor inlet pressure
R, r	Radius
S	Standard; usually used as a subscript
T, t	Temperature
T, t	Test condition or total when used as a subscript
T_{T_2}	Total compressor inlet temperature
T_{T_5}, T_5	Turbine inlet temperature
V_C, V_{CAL}, V_{cal}	Calibrated airspeed
V_I, V_{IND}, V_{ind}	Indicated airspeed corrected for instrument error
V_T, V_{TRUE}, V_{true}	True airspeed
V_{NE}	Never exceed airspeed
V_{99}	High side airspeed to achieve a 99-percent maximum specific range
V_{REC}, V_{rec}	Recommended cruise speed
W	Weight; measured in pounds
W_f	Fuel flow rate; usually in pounds per hour
%	Percent, parts per hundred
°	Degrees (units of temperature; units of angle)
Δ (delta)	Increment or change
ΔC_P	Excess power coefficient
δ (delta)	Ratio of an absolute pressure of interest to sea level standard absolute pressure
θ (theta)	Ratio of an absolute temperature of interest to sea level standard absolute temperature
μ (mu)	Advance ratio; ratio of helicopter forward speed to average rotational tip speed

Symbol	Definition
ρ (rho)	Air density; usually in slugs per cubic foot
Σ (sigma)	Summation
σ (sigma)	Ratio of an air density of interest to sea level standard air density
ω, Ω (omega)	Angular velocity; usually in radians per second

Abbreviation	Definition
A/C	Aircraft
aft	After, toward the rear
AMB, amb	Ambient; surrounding; as ambient air temperature
AVG, avg	Average
CAL, cal	Calibrated
CG, cg	Center of gravity; when used without prefix, usually refers to aircraft longitudinal center of gravity
config	Configuration; usually aircraft external configuration
CAS, cas	Calibrated airspeed
EGT	Exhaust gas temperature; usually in °C
eng	Engine
FAT	Free air temperature; temperature of ambient air
FPM, fpm	Feet per minute; a unit of velocity
FPS, fps	Feet per second; a unit of velocity
ft-lb	Foot-pound; a unit of torque or energy
ft/min, fpm	Feet per minute; a unit of velocity
FWD, fwd	Forward; toward the front

Abbreviation	Definition
G, g	Gravity unit; a unit of acceleration
GRWT, grwt	Gross weight; all-inclusive weight, pounds
IAS	Indicated airspeed; corrected for instrument error
ID	Inside diameter
IGE	In ground effect
ind	Indicated
in. Hg	Inches of mercury; a unit of pressure
in. H_2O	Inches of water; a unit of pressure
KCAS	Knots calibrated airspeed
KIAS	Knots indicated airspeed
KTAS	Knots true airspeed
KT, kt	Knot, knots; a unit of velocity
LAT, lat	Lateral
lat cg	Lateral center of gravity station
LONG., long.	Longitudinal
long. cg	Longitudinal center of gravity station
NAMPP	Nautical air miles per pound of fuel used; a unit of specific range
NAMI	Nautical air miles traveled; a unit of distance traveled
N_1	Rotational speed of gas producer train (compressor, gas producer turbine)
N_2	Rotational speed of power train (power turbine or output shaft)
MID, mid	Middle

Abbreviation	Definition
MP	Military power
MRP	Military rated power
OAT	Outside air temperature; ambient air temperature
OD	Outside diameter
OGE	Out of ground effect
para, paras	Paragraph, paragraphs
PSI, psi	Pounds per square inch; a unit of pressure
R/C	Rate of climb
R/D	Rate of descent
ref	Reference; material listed in an appendix but not included in this report
SC	Service ceiling; that altitude where rate of climb equals 100 feet per minute
SHP, shp	Shaft horsepower; usually engine output shaft horsepower
SL	Sea level; zero or reference on every altitude scale
S/N	Serial number
S/S	Sideslip or sideslip angle
STD, std	Standard; refers to standardized value or standard atmosphere; also used as a subscript
TAS	True airspeed
T/C	Time to climb
T/R	Tail rotor
TEMP, temp	Temperature

UNCLASSIFIED
Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) US ARMY AVIATION SYSTEMS TEST ACTIVITY EDWARDS AIR FORCE BASE, CALIFORNIA 93523	2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED 2b. GROUP

3. REPORT TITLE

ENGINEERING FLIGHT TEST,
YUH-1H HELICOPTER, PHASE D (LIMITED)

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

February 1966 through August 1967

5. AUTHOR(S) (First name, middle initial, last name)

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6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
NOVEMBER 1970	247	18

8a. CONTRACT OR GRANT NO. b. PROJECT NO. USAAVSCOM Project No. 66-04 c. USAASTA Project No. 66-04 d.	9a. ORIGINATOR'S REPORT NUMBER(S) USAASTA Project No. 66-04 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) N/A

10. DISTRIBUTION STATEMENT

This document may be further distributed by any holder only with specific prior approval of the Commanding General, USAAVSCOM, ATTN: AMSAV-R-F, PO Box 209, St. Louis, Missouri 63166.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY US ARMY AVIATION SYSTEMS COMMAND ATTN: AMSAV-R-F PO BOX 209, ST. LOUIS, MISSOURI 631[illegible]

13. ABSTRACT

This report presents the results of a limited Phase D Engineering Flight Test of the YUH-1H helicopter equipped with the Lycoming T53-L-13 engine. The objectives of the test were to define the increased performance, to evaluate the engine characteristics and to investigate the flying qualities resulting from the expanded flight envelope. One hundred and fifteen data flights were conducted from December 1966 to August 1967 accounting for 98 productive flight hours. Testing was conducted in California at Edwards Air Force Base and at auxiliary test sites at Bakersfield and Bishop. Helicopter performance was quantitatively defined for hover, takeoff, climb and level flight. Engine characteristics and helicopter flying qualities were determined quantitatively at specific flight conditions and were qualitatively investigated throughout the operating flight envelope. The performance of the UH-1D was significantly improved by the installation of the higher powered T53-L-13 engine which resulted in an increased weight, temperature and altitude flight envelope. The T53-L-13 engine demonstrated good dynamic characteristics except for instability at topping power and poor static characteristics. In general, the helicopter flying qualities were acceptable in forward flight. During low-speed flight and hover, the left directional control margin was inadequate. At a forward center of gravity, the helicopter has insufficient aft longitudinal control. To safely use the increased flight envelope, the control deficiencies must be corrected.

DD FORM 1 NOV 65 1473

UNCLASSIFIED
Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Limited Phase D Engineering Flight Test YUH-1H helicopter Lycoming T-53-L-13 engine						